

# JET PROPULSION

Journal of the  
AMERICAN ROCKET SOCIETY

*Rocketry . . . . Jet Propulsion Sciences . . . . Astronautics*

VOLUME 27

JUNE 1957

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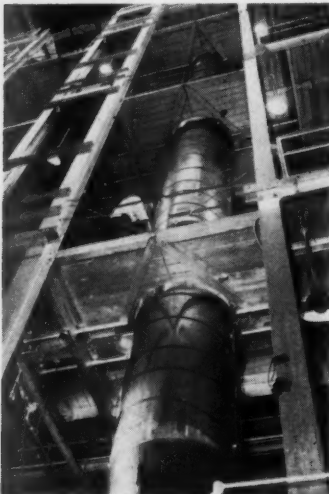
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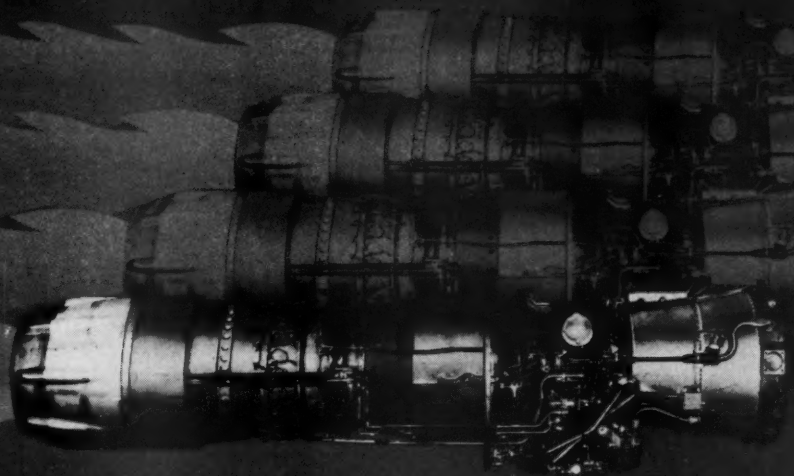
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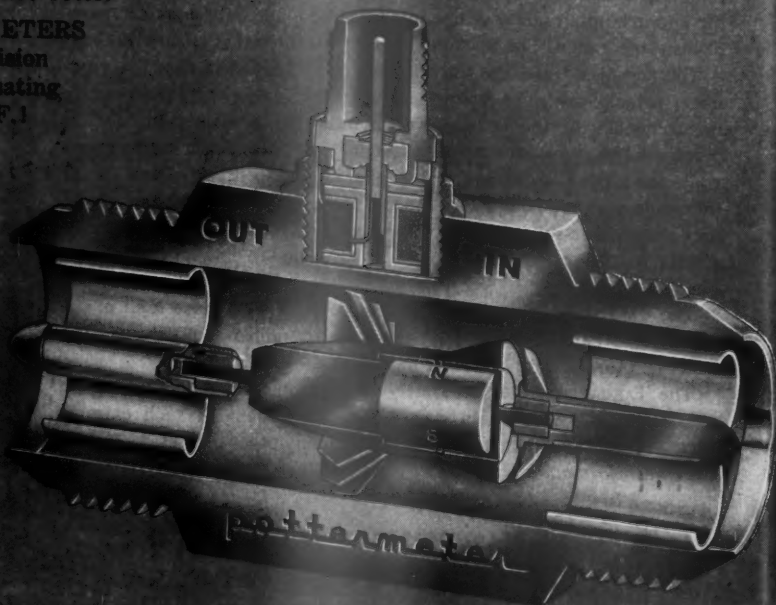
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## Scope of JET PROPULSION

JET PROPULSION, the Journal of the American Rocket Society, is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct; and thus it includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. JET PROPULSION is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. JET PROPULSION endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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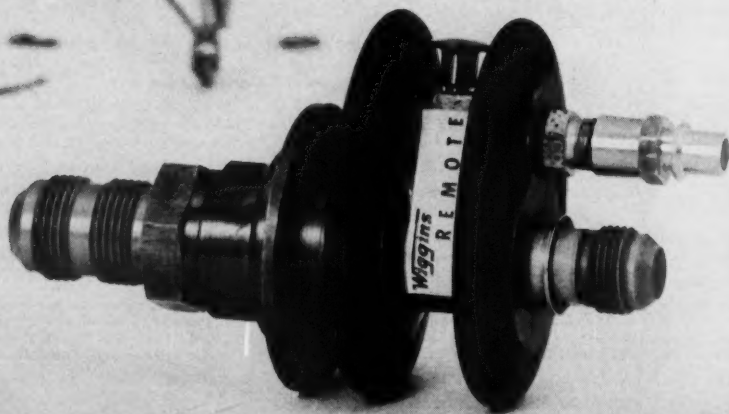
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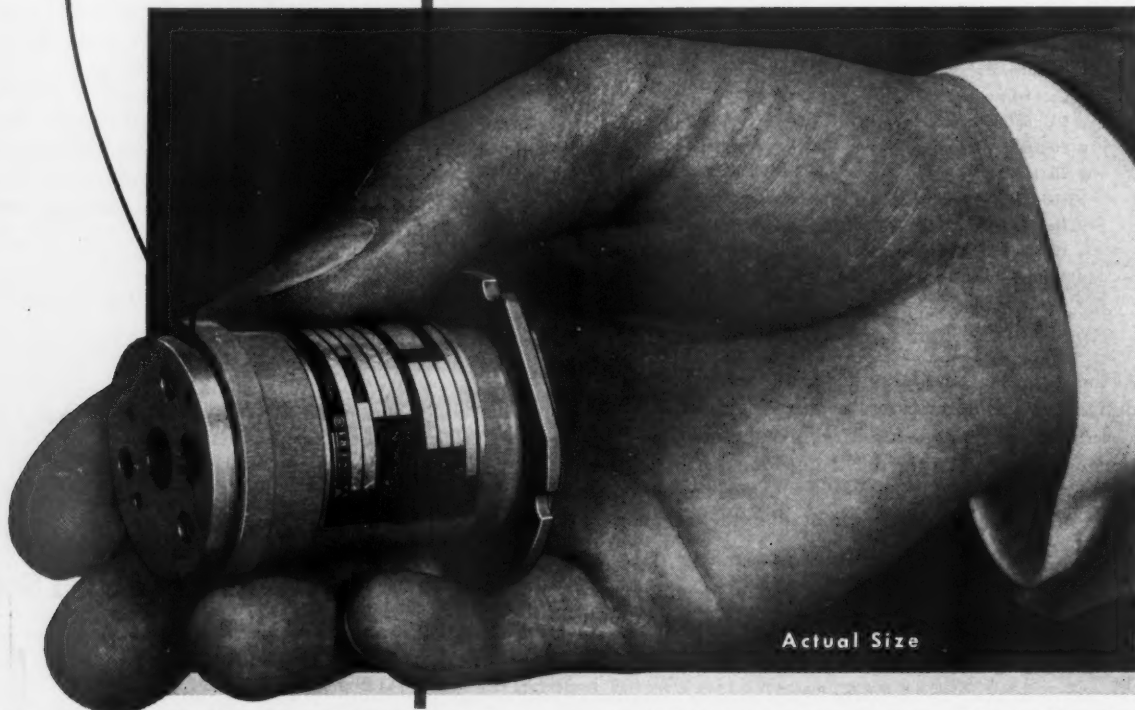
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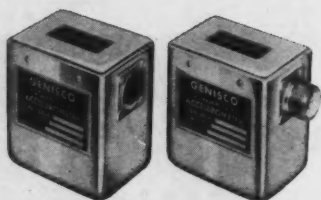
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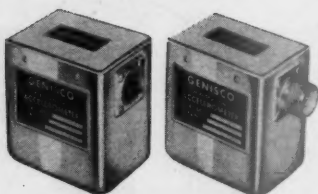
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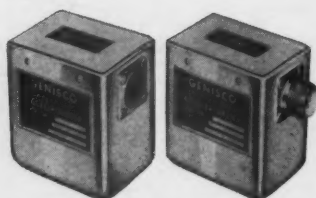
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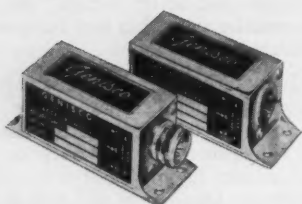
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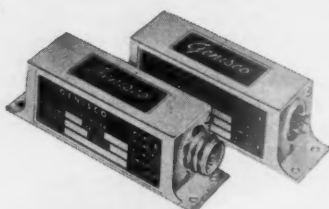
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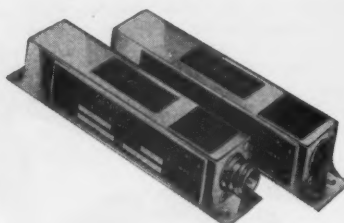
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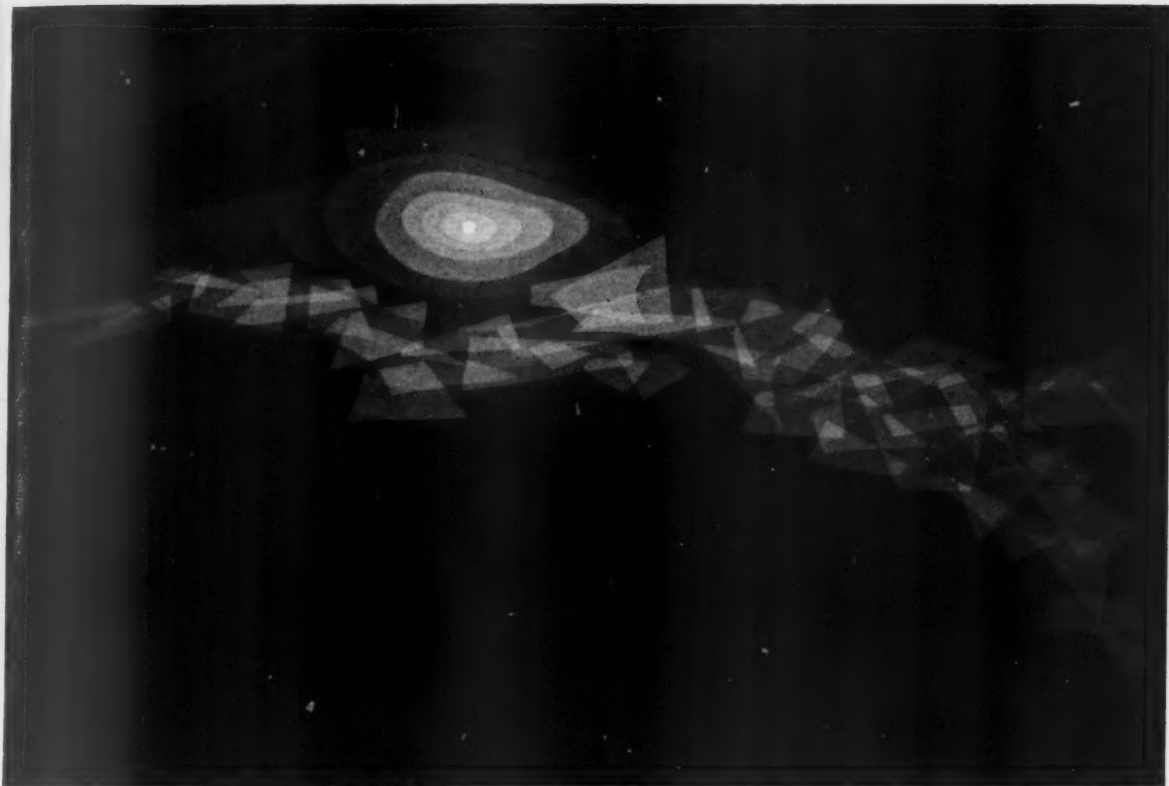
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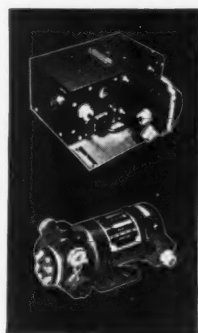
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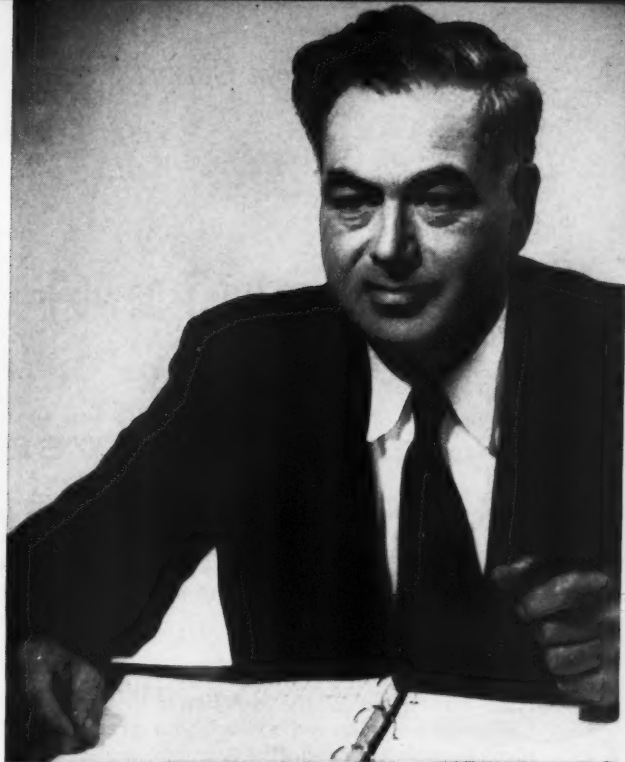
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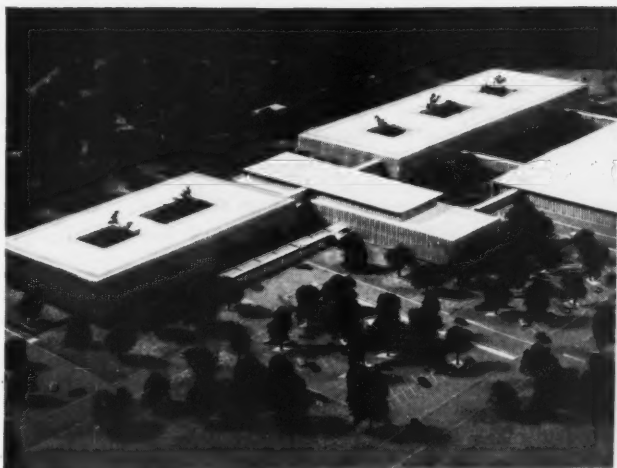
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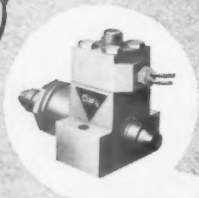
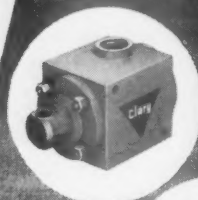
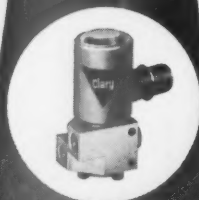
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# Reliability Concepts in Rocket Power Controls Design

H. L. COPLEN, JR.<sup>1</sup>

Aerojet-General Corp., Sacramento, Calif.

The principal power control problems in a liquid propellant rocket are propellant ignition, regulation during the thrust increase to rated thrust, malfunction detection during the starting transient, thrust control (if required), and mixture ratio or propellant-utilization control. The design of a rocket power control system must be considered an inseparable part of the general engine system design if an optimum control system is to result. Maximum engine reliability can be achieved only by a control system having functional simplicity as a result of real compatibility of the major engine components and the use of all inherent self-controlling elements in the system. The elimination of "service" systems, whether electrical, pneumatic, or hydraulic, in favor of direct actuation and sequencing from basic engine parameters is recommended as an approach to basic reliability. Other approaches to reliability through simplicity are the use of mechanical interconnection of valves for actuation, thermal isolation to avoid heater elements, and, in the case of missile engines, the location of malfunction and sequence monitoring equipment in the ground control console rather than the engine.

## Introduction

A HIGH degree of propulsion system reliability for missile use is exceeded in importance only by the basic requirement for an efficient basic engine design. These goals can be reached by designs achieving maximum compatibility of major engine components, simplicity, and functional redundancy of the control system. A liquid propellant rocket is a relatively complex hydromechanical and thermochemical power plant. Accordingly, the design goal is to seek control systems whose reliability derives from reduced numbers of operating components and the extensive use of simple mechanical hydraulic controls.

The basic types of liquid propellant rockets may be classified by propellant constituents used (i.e., monopropellant, bipropellant, etc.), the type of propellant pressurization system, and the propellant reactivity or ignition problem of a particular bipropellant system. A propellant system is said to be auto-ignitable if the fuel and oxidizer combinations react sufficiently rapidly to self-ignite themselves with an acceptably short ignition delay. Only a few of the currently used fuel and oxidizer combinations are auto-ignitable. Because of logistic problems, the auto-ignitable fuels are not generally used. In some cases, these fuels are used in small quantities as an additive or starting fluid, but the currently used propellant combinations are usually not auto-ignitable.

<sup>1</sup> Presented at the ARS 11th Annual Meeting, New York, N. Y., Nov. 26-29, 1956.

<sup>1</sup> Principal Engineer, Systems and Controls Dept., Liquid Rocket Plant. Mem. ARS.

The engines whose control problems are discussed herein will be the bipropellant types, since this field is of prime current interest.

A simple re-operative, bipropellant rocket engine having a gas-pressurized propellant-feed system is shown in Fig. 1. Regulation of this type of unit is generally confined to transient start regulation based on a timed opening of the propellant control valve. The thrust level is established by the pressurization regulator setting.

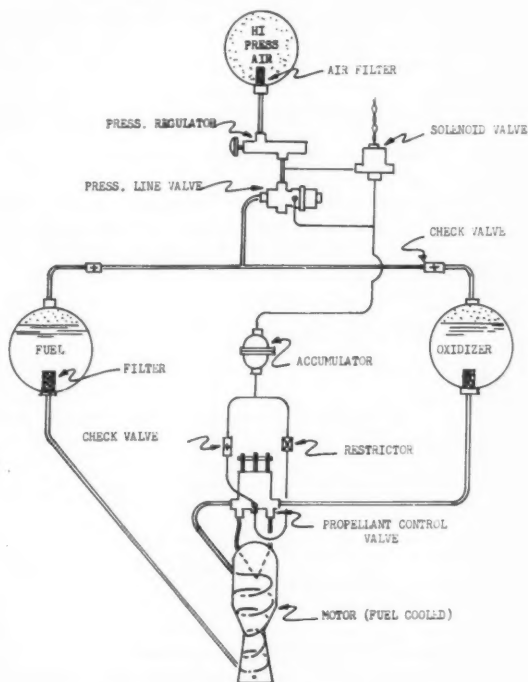


Fig. 1 Simple re-operative bipropellant system

A turborocket system generally similar to that of the V-2 is shown in Fig. 2. Thrust regulation for this system consisted of a constant flow regulation device in the monopropellant flow line to the gas generator (hydrogen peroxide as illustrated here).

A typical regenerative turborocket system is shown in Fig. 3. This type unit may have the turbopump started by an auxiliary cold gas supply to the turbine nozzle followed by ignition and "bootstrapping" to full thrust.

Some of the propellants for well-known bipropellant systems are given in Table 1. Typical performance values for some combinations of these propellants are given in Table 2.

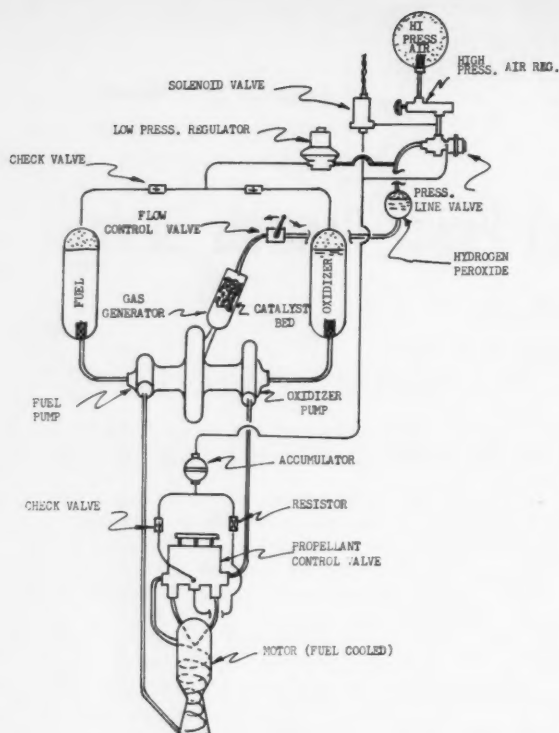


Fig. 2 Turbopropellant system (gas turbine operated by hydrogen peroxide)

Currently, the propellant systems receiving most serious consideration for large missiles and aircraft super-performance power plants are the hydrocarbon fuels and liquid oxygen or white fuming nitric acid.

### Principal Power Control Problems

There are several major control requirements in a rocket engine power plant which vary in significance depending on the specific propellant system adopted, the type of propellant pressurization employed, and the application intended. Since the relative importance of these requirements may not be evident to all, a brief discussion of each is believed pertinent to this paper.

The most critical problem on any liquid propellant rocket is the control of ignition and combustion malfunctions during the starting transient. Starting reliabilities are typically as low as 0.60 for a rocket with a running reliability of 0.95 to 1.00. A rocket propellant system inherently handles extremely high energy densities using a liquid oxidizer rather than air. Consequently, the propellant-mass flow rates are astronomical compared to the fuel flow rates of air-consuming engines. The high power density in the combustion chambers makes the establishment of stable combustion and protection against unburned propellant accumulation in these chambers an absolute necessity to safety in piloted aircraft and to reliability in missile propulsion systems. The basic work to establish safety requirements and apply workable controls to current engines has been in progress for some time at Aerojet-General Corp. The necessity for physical and functional integration of the malfunction-monitoring features with the propellant injection and flow control means of the combustion chamber was emphasized as a result of this work. In addition, both rapid response speed and reproducibility of operation are necessary to effective control.

Ignition reliability is most readily achieved by the use of

a multiplicity of ignition heat sources—several solid propellant igniters, a separate auto-ignitable propellant supply, or a generous-capacity spark source. In most engines, this can be readily achieved. However, direct monitoring of the igniter performance is essential to avoid dangerous propellant accumulation in the thrust chamber coincident with possible auto-ignition.

Transient start sequencing requirements may be handled as a function of time or as a function of turbopump speed or propellant pump discharge pressures. Use of the latter two parameters for sequencing is a logical approach since it inherently monitors some types of malfunction. Other types of malfunctions, such as flameouts and combustion instability, must be monitored independently.

Current combustion-malfunction control designs for aircraft rockets are based on the hydromechanical feedback of pressures. The instantaneous behavior of the combustion process directly actuates an injector propellant controller. Thus all unnecessary delays are eliminated. In this way, the initial ignition process as well as the subsequent transient combustion conditions may be continuously monitored. With missile rocket power plants, the malfunction-monitoring device is preferably located in the launcher control console to reduce engine complexity and weight.

The control of thrust is the next most important control problem. As contrasted to the high frequency response requirements of the malfunction-control system, the thrust control system must be limited in frequency response to about 10 cps or less. The purpose is to prevent spurious pressure transients from initiating regenerative engine-pressure oscillations. The control accuracy requirements are again a function of application; most ATO and super-performance installations would not require a high degree of thrust repeatability—perhaps not better than  $\pm 5$  per cent for a fixed-thrust rocket or for the limits of a variable thrust rocket. However, some missile applications require accurate thrust level control and accurate thrust cutoff reproducibility because of the guidance systems adopted. Accordingly, the system and feedback source adopted will vary with the engine system and must be specifically tailored to it.

Several feedback sources have been considered but only a few have been utilized in actual rocket designs. These are direct thrust, chamber pressure, one or both of the propellant feed pressures, and turbopump rotative speed. Direct-thrust measurements require the use of complex thrust chamber mounting structures and thrust transducer; these are usually unjustifiably heavy. Chamber pressure is a desirable selection but requires high temperature gas source connections and does not inherently provide starting-transient regulation. Fuel or oxidizer feed pressure feedback provides a reasonable through indirect measurement of chamber pressure and, in addition, usually provides for the starting-transient regulation. Turbopump speed is useful only as a secondary feedback for applications requiring power equalization between several rocket engines or their separate pumping plants. Accordingly, one of the first three primary feedback sources would be selected to suit the intended application.

Large volumes of high pressure fuel are often available for use as a control-system actuation fluid; this power source has been seriously considered in recent designs. Required actu-

Table 1 Rocket propellants of current interest

Fuels	Oxidizers
JP-4 (jet fuel)	WFNA (white fuming nitric acid)
RP-1 (kerosene)	RFNA (red fuming nitric acid)
Ammonia	RFNA with additives for corrosion inhibition
Hydrazine	liquid oxygen
Liquid hydrogen	liquid fluorine

Table 2 Performance of selected bipropellant rocket systems

Fuel	Oxidizer	$MR W_{oz}/W_f$	$I_{sp}$	$P_{c_0}$ assumed	Reference <sup>1</sup>
JP-4	WFNA	4.7	299	300	(1)
"	RFNA				
"	LOX	2.3	287	650	(2)
RP-1	LOX	2.3	286	650	
Hydrazine	WFNA	1.22	246	300	(3)
"	RFNA	1.56	245	300	(4)
"	LOX	1.0	268	300	(2)
"	LF1	2.16	294	300	(3)
Ammonia	WFNA	3.0	232	300	(5)
"	RFNA				
"	LOX	1.4	260	300	(2)
"	LF1	3.35	311	300	(6)
Liquid hydrogen	LOX	3.6	348	300	(4)
"	LF1	5.85	367	300	(4)
"	Ozone	2.4	386	300	(3)

<sup>1</sup> Data obtained from the following sources: (1) "Rocket Data for Rocket Engines," Bell Aircraft Corp., Jan., 1954. (2) Unpublished calculations by Aerojet-General Corp. personnel. (3) Project Rand, "A Compilation of Computed Specific Impulse Values," RA-15049, Douglas Aircraft Co., Inc., 1947 (confidential). (4) A. L. Lemmon, "Fuel Systems for Jet Propulsion," OSRD, 1945 (confidential). (5) Aerojet Report RTM-58, "Thermochemical Calculations on the WFNA and JP-3 Propellant Combination," April 5, 1950 (confidential). (6) Morrell, V. E., "Effect of Combustion Chamber Pressure and Nozzle Expansion Ratio on Theoretical Performance of Several Rocket Propellant Systems," NACA RM, RM E50c30, May 25, 1950 (confidential).

ation power levels are usually high because of the need for large propellant throttling valves operating at high pressure and high speeds. In addition, electrical power requirements for control purposes must be kept at a minimum for most rocket applications. The use of high pressure air or gases for actuation, purge, or regulation purposes is minimized to eliminate the pressure regulation systems required or to minimize the size of the high pressure storage vessels.

The control of propellant mixture ratio has been accomplished by the calibration of the engine hydraulic systems and empirical adjustment of system resistances to correct for production tolerances. This simple means can be used to limit mixture ratio variations to within the limits of thrust chamber tolerance for most current designs. The alternative of metered mixture ratio control is not consistent with engine simplicity and is not actually necessary.

For ATO and super-performance applications, fuel is usually taken from the main aircraft fuel system, and neither thrust nor duration are critical. Accordingly, a metered mixture ratio control in these applications does not appear to be warranted.

For missile applications, mixture ratio control is intended to assure maximum total impulse by assuring the simultaneous exhaustion of both propellant constituents. Since variations in actual propellant tankage will occur due to production and propellant servicing tolerances, the use of a constant mixture ratio control would be unsuitable. Rather, a mixture ratio trim device based on a tank level ratio measurement is required to adjust the operating ratio and thus effect the simultaneous exhaustion of propellant supplies.

The development of shutdown reproducibility (i.e., consistency) of total impulse supplied after command shutdown signal is frequently required. This requires a control system with minimum sequential operations and with maximum repeatability response in the principal components. This reliability must be achieved without developing excessive line pressures resulting from water hammer in the propellant system.

### Inherent Reliability in Controls Design

The most reliable engine system must contain the fewest possible operating elements; these components must individually have been developed to their highest state of individual reliability. Such an engine could be said to have "inherent" reliability. Unfortunately, this goal is elusive

because contractual requirements for meeting a schedule are usually overriding considerations. Thus, a basic incompatibility in system elements or a newly discovered control requirement is frequently solved by the introduction of additional control functions or devices. A basically more unreliable engine system will result and no practical degree of individual component reliability can fully overcome this. However, the reliability of the individual control components is a most important consideration.

Because of the very large investment of time and funds represented by the initial rocket engine hardware and the high cost of conducting rocket test firings, it is essential that the functional requirements for the power control system elements be accurately predetermined mathematically and that the experimental parts be bench-tested and adjusted prior to the rocket firing test. Analog and digital computers have been shown to be of great value in facilitating the development of rocket controls, particularly those for regenerative rocket systems. They are standard design tools at Aerojet-General and have been used to establish the following:

Major component design criteria for compatibility and controllability.

Basic system stability criteria.

Starting transient performance of "bootstrapping" systems.

Transient hydrodynamics of engine propellant systems.

Performance error bands due to operational tolerances.

The detail functional design requirements for control components.

Thus, the optimum system parameters and the control parameters for a specific application may be predetermined. The optimum controller gains may be established and the controls bench-adjusted so that firing tests are initially acceptable and need not be used to adjust the power controls empirically.

The basic control design policies in use at Aerojet-General can be summarized as follows:

### Direct Mechanical Coupling of Devices of Similar Function

This is applicable to the pump drive assembly, the thrust chamber propellant control valves, and the gas generator propellant control valves. A system so designed has inherent mixture ratio regulation without complicated controls. Most rocket pump-drive assemblies today are built with both the fuel and oxidizer pump driven directly from a single turbine. However, mechanical interconnection of propellant-control

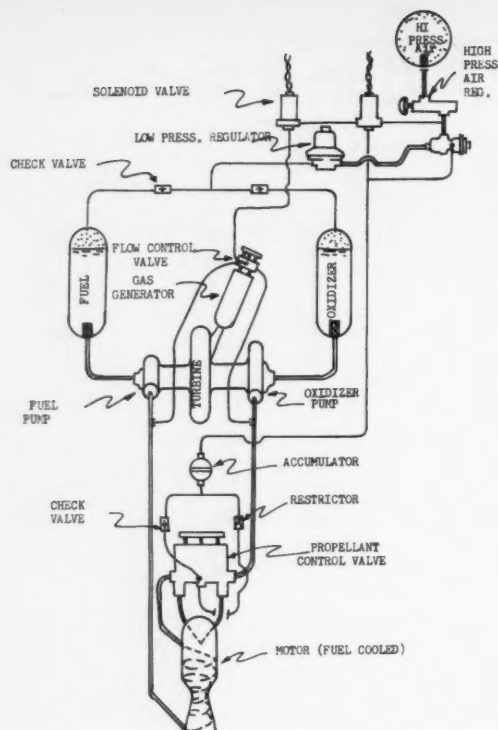


Fig. 3 Turborocket system (gas turbine regeneratively operated)

valves to utilize a single actuator and pilot valve is somewhat less universal.

#### Elimination of Auxiliary Control Power Systems

Such systems generally use regulated nitrogen or helium pressure, regulated hydraulic pressure, electrical regulation and actuation, and sequencing power supply systems. All of these systems are really extraneous to the rocket engine system. The intent is to largely eliminate these auxiliary power systems with their attendant complexity by the maximum use of the fuel-pressure system as a source of actuation fluid. Some weight penalty must be paid in the actuator size, but this is more than offset by the weight eliminated from the hydraulic accumulator or pneumatic pressure storage vessels thus replaced.

#### Start Sequencing from System Parameters

The starting sequence of a rocket engine can be designed around a time base, or a propellant availability criterion. A basic system parameter must be used as an index; such an index is propellant pressure or, for convenience, fuel-pump discharge pressure. The latter technique is attractive in that it eliminates the use of timers with arbitrary time bases. A timer-monitored start sequence may actually be incompatible with engine operation over a wide range of environmental conditions. The pressure-sequencing method, with pump-drive assembly acceleration rate as the real time base, would at first seem to be quite variable. However, the system may be designed to utilize the portion of the starting transient where the turbopump torque-to-load (principally inertia) ratio is very reproducible from run to run as well as engine to engine. Accordingly, the reproducibility can be shown to be adequate for most applications.

#### Relocation of all Engine Controls Possible in Launcher System

With a missile engine, the desire is to ensure a full-duration engine run at all costs once the missile has been launched.

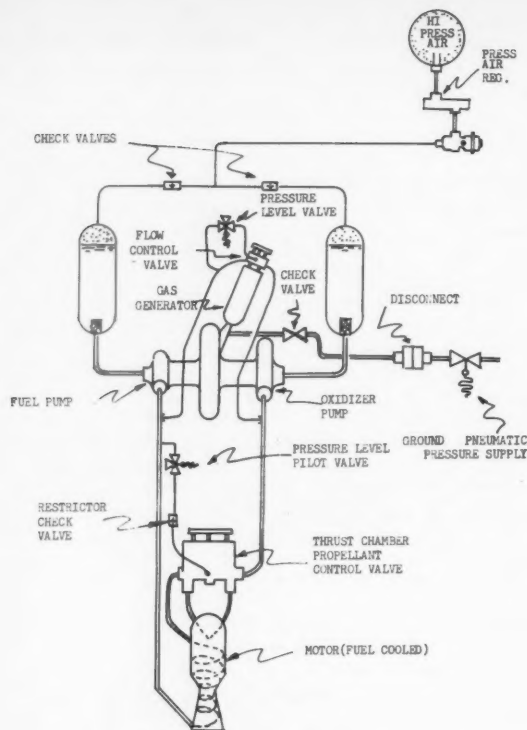


Fig. 4 Self-actuated missile turborocket system

It is evident that all malfunction monitoring and sequence controls which could shut the engine down should be disconnected after launching and therefore should be located in the launcher control system. In this category fall the ground ignition systems and starting propellant systems, ignition, speed and temperature-monitoring controls, and missile-release controls. In-flight controls are limited to corrective devices rather than shutdown controls and no automatic engine-malfunction shutdown provisions are made. Command shutdown provisions are made for exclusive use of the automatic guidance and remote ground-command signal channels.

A simple self-actuated control system is shown in Fig. 4 for comparison with Fig. 3. The system shown uses a ground-supplied pneumatic pressure supply for accelerating the pump-drive assembly to a "bootstrap" or regenerative pressure level.

#### Reliability Development and Demonstration

Reliability development begins in the functional system design where a system with minimum complexity is conceived and developed in design to have a maximum number of redundant features without an unnecessary physical duplication of functions. Next, the selection of optimum control means and the design of components with an ideal combination of physical and functional integration must be completed prior to actual hardware development. This design effort must be supported by intensive materials and applications development if it is to be successful.

Very thorough testing for functional performance of control hardware under the extreme environmental conditions of vibration, temperature, and fluid exposure is necessary to achieve a suitable reliability level for a component. The early concept of a cheap, one-shot component for either an

(Continued on page 643)

# On the Powered Flight Trajectory of an Earth Satellite

BURTON D. FRIED<sup>1</sup>

The Ramo-Wooldridge Corp., Los Angeles, Calif.

The problem of programing the powered flight trajectory of an earth satellite to obtain maximum orbit altitude is investigated. For a point mass missile, moving in a uniform gravitational field with aerodynamic forces neglected, the optimum trajectory is obtained by making the tangent of the angle  $\psi$  between the thrust vector and the horizontal a linear function of time. This result is valid for arbitrary time dependence of thrust and mass, including the discontinuities corresponding to staging. Moreover, if we replace orbit altitude by some other function of the burnout altitude and velocity vector, e.g., the altitude at perigee for an elliptical orbit, and look for a stationary solution, we again obtain a linear time dependence for  $\tan \psi$ . Explicit expressions for velocity and altitude with this type of thrust attitude program are given for the particular case of constant thrust and mass flow rate.

## Introduction

THE actual choice of a powered flight trajectory for the forthcoming IGY satellite<sup>2</sup> will depend upon details of booster design and upon the particular orbit selected, the latter being determined by instrumentation and data requirements. Independent of these considerations, however, it is of interest to ask what type of trajectory will be optimal from the standpoint of missile efficiency.

To begin with we must select some criterion of excellence for comparing trajectories. Whereas range is an obvious choice for surface to surface missiles, it is not clear what quantity will play a similar role for satellites. One important property of a satellite is its useful orbiting lifetime. Too low an orbit altitude will result in a rapid slowing down due to atmospheric drag and an undesirably short lifetime. This is not an important consideration if the booster is sufficiently powerful, for at altitudes above about 300 miles the orbiting lifetime is measured in years and it is likely that the power supply life rather than the drag induced decay would be the determining factor. However, if the performance of the booster vehicle is marginal, orbit altitude will be of crucial importance and can serve as the desired criterion.

The orbit "altitude" is a well defined quantity only for circular orbits. In the case of elliptical orbits it is convenient to work with the distance of closest approach (d.c.a.), i.e., the altitude of the satellite at perigee. This is approximately correlated with the drag induced decay, since the major part of the slowing down will occur near perigee. The correlation is not exact, for of two orbits with equal d.c.a. the one with the greater eccentricity will have the longer life. However, the analysis described below would not be changed essentially if the d.c.a. were replaced by, say, the mean of the altitudes at perigee and apogee or by some other function of the altitude and vector velocity at the end of powered flight. For definiteness, we shall adopt the d.c.a. as our criterion.

Analytical results on trajectories can only be obtained by neglecting certain terms in the equations of motion, for if we include atmospheric effects, variations in the gravitational force

field, etc., then solutions of the differential equations can only be obtained by numerical integration. It is rather gratifying to find that with a few, not unreasonable approximations we are led to a very simple result: The tangent of the thrust attitude angle should be a linear function of time.<sup>3</sup> In the following section the derivation of this result and the method of determining the coefficients in the linear relation will be presented. We shall see that while this result is an approximate one, it can be of practical value when correctly applied.

Our approach will be to assume that the characteristics of the booster, i.e., its thrust and mass as functions of time, are specified. The problem is then to determine what powered flight trajectory will put the satellite into an orbit of maximum altitude. Many alternative formulations are possible, e.g., for given payload (i.e., orbiting weight) and orbit, find the trajectory which minimizes the take-off weight. Problems of this sort, however, require fairly detailed information about the booster—staging configuration, structural efficiencies, tankage factors, etc.—so that solutions of general validity can scarcely be obtained. These difficulties are avoided when, as here, the properties of the missile are regarded as given.

## Trajectory Analysis

Consider the problem of finding the powered flight trajectory which will result in maximum satellite orbit altitude for a missile whose thrust magnitude and mass are specified, albeit arbitrary,<sup>4</sup> functions of time,  $F(t)$  and  $M(t)$ . We shall treat the missile as a point mass, the earth as a nonrotating sphere, and to begin with shall neglect three things: (i) Aerodynamic effects—drag, lift, etc., (ii) the dependence of thrust on altitude, (iii) the variation of the magnitude and direction of gravity during powered flight.

If the orbit is to be circular (elliptical orbits are discussed later) than at booster burnout the vertical component of velocity must be zero and the tangential component must have such a value that the centripetal and gravitational forces just balance

$$u_x = u_{\text{sat}} \equiv (gR_e)^{1/2}(1 + y/R_e)^{-1/2}, \quad u_y = 0 \dots [1]$$

where  $(u_x, u_y)$  is the booster velocity at burnout,  $R_e$  is the earth's radius,  $y$  is the burnout altitude and  $g$  is the acceleration of gravity at the earth's surface. For planar trajectories, the path is completely determined by specification of the thrust attitude angle (say with the horizontal)  $\psi(t)$ . We then ask what function  $\psi(t)$  maximizes the altitude at burnout subject to the conditions [1]. For simplicity, we shall first assume  $u_{\text{sat}}$  to have a given value, ignoring its dependence on altitude (which will anyhow be slight for altitudes small compared to the radius of the earth). Later we shall see that this dependence can easily be taken into account.

<sup>3</sup> The stationary character of this thrust attitude program for circular orbits has also been demonstrated by Perkins, F. M., "Flight Mechanics of Ascending Satellite Vehicles," JET PROPULSION, vol. 26, May 1956, pp. 352-358.

<sup>4</sup> The term "arbitrary function" is to be understood in a physical rather than a mathematical sense, i.e., as a function which is arbitrary except for such restrictions of continuity, differentiability, positive definiteness, etc., as may be necessary to exclude situations which are clearly not sensible in view of the physical significance of the function.

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<sup>1</sup> Member of the Technical Staff, Aeronautical Research Laboratory.

<sup>2</sup> See Newell, H. E., "Scientific Uses of an Artificial Earth Satellite," JET PROPULSION, vol. 25, Dec. 1955, pp. 712-715.

If  $\delta y$  is the change in burnout altitude due to a change  $\psi(t) \rightarrow \psi(t) + \delta\psi(t)$ , with similar definitions for  $\delta u_x$  and  $\delta u_y$ , then a necessary condition for  $y$  to be a maximum subject to [1] is

$$\delta y + \lambda \delta u_x + \mu \delta u_y = 0 \dots\dots\dots [2]$$

where the Lagrange multipliers  $\lambda, \mu$  are unknown constants. If

$$N(t) = F(t)/M(t)$$

is the instantaneous thrust to mass ratio, we have

$$\left. \begin{aligned} u_x &= \int_0^T dt N \cos \psi + u_x^0 \\ u_y &= \int_0^T dt N \sin \psi - gT + u_y^0 \\ y &= \int_0^T dt (T-t)N \sin \psi - \frac{1}{2}gT^2 + y^0 + u_y^0 T \end{aligned} \right\} \dots [3]$$

where  $T$  is the total burning time. If  $\psi(t) \rightarrow \psi(t) + \delta\psi(t)$ , then

$$u_x \rightarrow u_x - \int_0^T dt N \sin \psi \delta\psi(t)$$

so

$$\delta u_x = - \int_0^T dt N \sin \psi \delta\psi$$

Similarly

$$\delta u_y = \int_0^T dt N \cos \psi \delta\psi$$

$$\delta y = \int_0^T dt (T-t)N \cos \psi \delta\psi$$

Upon substituting these expressions into [2], we see that [2] will be satisfied for arbitrary  $\delta\psi(t)$  if and only if

$$N[(T-t) \cos \psi - \lambda \sin \psi + \mu \cos \psi] = 0$$

for all  $t$ . Thus, when  $N \neq 0$  we require

$$\tan \psi = \lambda^{-1}[(\mu + T) - t] \dots\dots\dots [5]$$

Since  $\lambda$  and  $\mu$  are anyhow unknown constants, we may simply state our result in the form

$$\tan \psi = a - bt \dots\dots\dots [6]$$

where  $a$  and  $b$  are constants which must be chosen to satisfy [1].

Although the result [6] is valid for arbitrary  $F(t)$  and  $M(t)$ , the explicit values of  $a$  and  $b$  will, of course, depend upon these functions. In the particular case where  $F(t)$  and  $\dot{M}(t)$  are (piecewise) constant functions, the expressions [3] for  $u_x$  and  $u_y$  can be integrated in terms of elementary functions. The increments in  $u_x$ ,  $u_y$  and  $y$  due to one stage of mass ratio  $r$ , exhaust velocity  $c$  and burning time  $t_1$ , when  $\tan \psi$  is a linear function of time are

$$\left. \begin{aligned} \Delta u_x &= c \cos \bar{\psi} \left\{ \ln r - \ln \left[ \frac{1 + \cos(\psi_0 - \bar{\psi})}{1 + \cos(\psi_1 - \bar{\psi})} \cdot \frac{\cos \psi_1}{\cos \psi_0} \right] \right\} \\ \Delta u_y &= \Delta u_x \tan \bar{\psi} + c \ln \left[ \frac{1 + \sin \psi_0}{1 + \sin \psi_1} \cdot \frac{\cos \psi_1}{\cos \psi_0} \right] \\ \Delta y &= t_1 \left[ c \frac{\cos \psi_1 - \cos \psi_0}{\sin(\psi_0 - \psi_1)} - \frac{\Delta u_y}{r-1} \right] \end{aligned} \right\} \dots [7]$$

where  $\psi_0$  is the initial value of  $\psi$ ,  $\psi_1$  is the value at time  $t_1$  and  $\bar{\psi}$  is the value at time

$$\bar{t} \equiv \frac{M_0}{\dot{M}} = \frac{t_1 r}{r-1}$$

$M_0$  and  $\dot{M}$  being the initial mass and the mass flow rate for the stage.<sup>5</sup> By summing the contributions of each stage and adding (to  $u_x$  and  $y$ ) the gravitational terms ( $-gT$  and  $-\frac{1}{2}gT^2$ , where  $T$  is the total burning time), we obtain expressions for  $u_x$ ,  $u_y$  and  $y$ . Setting  $u_x = u_{\text{ast}}$  and  $u_y = 0$ , we would use two of these relations to find  $a$  and  $b$  and then substitute the resulting values of  $a$  and  $b$  into the expression for  $y$  to find the orbit altitude.

So far we have only demonstrated that [6] provides a stationary solution. However, it is easy to show that the altitude obtained with this thrust attitude program is actually an absolute maximum. The proof, which is due to G. Culler and K. Hoffman, is given in the Appendix.

While the actual process of determining  $a$  and  $b$  is straightforward, it is somewhat lengthy in practice since it involves the solution of two simultaneous transcendental equations. Also, our derivation has neglected the dependence of  $u_{\text{ast}}$  upon altitude. Both of these difficulties can be resolved if we adopt a somewhat more general approach.

We drop the previous restriction to circular orbits and simply assume that at burnout we have certain values of  $u_x$ ,  $u_y$  and  $y$ . These uniquely determine the Kepler ellipse which the satellite follows and, in particular, the distance of closest approach (d.c.a.) of this orbit. If  $H$  is the d.c.a. and

$$D = R_e + H$$

then in terms of energy  $E$  and angular momentum  $L$  we have

$$D = \frac{-k + (k^2 + 2EL^2)^{1/2}}{2E} \dots\dots\dots [8]$$

where

$$\left. \begin{aligned} E &= \frac{u^2}{2} - k(R_e + y)^{-1} \\ L &= u_x(R_e + y) \\ k &= gR_e^2 \end{aligned} \right\} \dots\dots\dots [9]$$

If  $\psi(t)$  is to maximize  $D$ , then we require

$$\delta D = D_y \delta y + D_{u_x} \delta u_x + D_{u_y} \delta u_y = 0 \dots\dots\dots [10]$$

where  $D_y = \partial D / \partial y$ , etc. Using [4] we again find

$$\tan \psi = a - bt$$

where  $a$  and  $b$  are now given by

$$a = (TD_y + D_{u_y})/D_{u_x} \quad b = D_y/D_{u_x} \dots\dots [11]$$

This provides a systematic, iterative procedure for calculating  $a$  and  $b$ : Guess  $u_x$ ,  $u_y$ ; calculate  $D_{u_x}$ ,  $D_{u_y}$ ,  $D_y$  from [8, 9]; compute  $a$  and  $b$  from [11]; find  $u_x$ ,  $y$ ,  $y$  from [7]; etc. This should yield the maximum<sup>6</sup>  $D$  possible with this missile, subject to the assumptions i, ii and iii.

Of these, the most serious is the first, since a program like [6] would result in intolerably large angles of attack. Consequently, it seems advisable to use a "gravity turn" (thrust parallel to velocity) until the missile is high enough to justify the neglect of aerodynamic forces, at which time we can switch over to the program [6]. Using the fact<sup>7</sup> that the velocity magnitude at the end of a gravity turn is a far less sensitive function of the initial conditions (kick angle, etc.) than is the velocity attitude, it is easy to show that the gravity

<sup>5</sup> In terms of the  $a$  and  $b$  of [6],  $\tan \psi_0 = a$ ,  $\tan \psi_1 = a - bt_1$ ,  $\tan \bar{\psi} = a - [(bt_1 r)/(r-1)]$ .

<sup>6</sup> The d.c.a. is stationary for the thrust attitude program [6], with  $a$  and  $b$  given by [11]. Although we have proved that it is actually a maximum only for the special case of circular orbits (cf. Appendix), arguments of physical continuity would suggest that, at least for orbits of small eccentricity, [6] also maximizes the d.c.a. of elliptical orbits.

<sup>7</sup> Culler, G., and Fried, Burton D., "Universal Gravity Turn Curves," *Journal of Applied Physics*, in press.

turn should be chosen so that at the time of transition to the regime [6],  $\psi$  has no discontinuity.

This program also takes care of assumption ii, since we follow [6] only at high altitudes where the thrust is independent of altitude. Finally, the fact that the powered flight takes place in a nonconstant gravitational force field would modify our results somewhat. However, this effect will be small provided the distance covered during powered flight is not large compared to the earth's radius.

## Conclusions and Discussion of Results

We have shown that to obtain a satellite orbit of maximum altitude with a given missile, the thrust attitude angle  $\psi$  should vary with the time according to

$$\tan \psi = a - bt \dots \dots \dots [6]$$

during that portion of the powered flight where aerodynamic forces and variations in gravity are unimportant. This result is valid for arbitrary time dependence of thrust and mass, and hence for arbitrary staging configurations. Moreover, if we replace orbit altitude by some other function of the burnout altitude and velocity vector, e.g., the altitude at perigee, and look for a stationary solution, we again obtain a result of the form [6].

The evaluation of  $a$  and  $b$  requires that we calculate the burnout velocity with  $\psi$  given by [6]. For the particular case where thrust and mass flow rate are constant, this can be carried through analytically. The results—Equations [7]—are a generalization of the familiar *clnr* formulas which are obtained when  $\psi$  is constant.

A steering program like [6] is physically reasonable. To obtain high altitudes, we would expect to start firing with large  $\psi$ , so that the resultant vertical velocity could, in the course of time, produce altitude. Later in the flight we would want smaller  $\psi$  in order to build up the required horizontal velocity and also to allow gravity to cancel any accumulated vertical velocity. The optimum way of combining these considerations is prescribed by [6]. Since the general form of this function is qualitatively similar to that associated with a gravity turn, we can expect that the penalty of using the latter during the atmospheric portion of the flight will not be large.

## APPENDIX

We shall demonstrate here that the altitude of a circular orbit is not only stationary for the thrust attitude program [5] but actually has an absolute maximum, as first pointed out by G. Culler and K. Hoffman (unpublished). We simply compute the second variation of the quantity

$$J[\psi] = (y + \lambda u_x + \mu u_y)$$

the vanishing of whose first variation leads to [5]. We have

$$\frac{\delta^2 J}{\delta \psi^2(t)} = -N[(\mu + T - t) \sin \psi + \lambda \cos \psi] \dots [A.1]$$

Let  $\psi_0(t)$  be the thrust attitude program defined by [5] or [6], i.e.,

$$\tan \psi_0(t) = a - bt = \lambda^{-1}(\mu + T - t), \quad 0 < \psi_0 < \pi/2 \dots [A.2]$$

Then [A.1] gives

$$\begin{aligned} \frac{\delta^2 J}{\delta \psi^2(t)} &= -N\lambda[\cos \psi + \tan \psi_0 \sin \psi] \\ &= -N\lambda \sec \psi_0 \cos(\psi - \psi_0) \dots \dots \dots [A.3] \end{aligned}$$

We see that provided the physically reasonable conditions  $0 < \psi_0 < \pi/2$  and  $\lambda > 0$  are satisfied<sup>a</sup> we have

$$\left. \frac{\delta^2 J}{\delta \psi^2(t)} \right|_{\psi = \psi_0(t)} < 0 \dots \dots \dots [A.4]$$

which guarantees that  $J$  has a relative maximum for  $\psi = \psi_0(t)$ . Since

$$\begin{aligned} J &= \int_0^T dt N \{ (T + \mu - t) \sin \psi + \lambda \cos \psi \} \\ &= N\lambda \sec \psi_0 \int_0^T dt \cos(\psi - \psi_0) \end{aligned}$$

we see that  $\psi = \psi_0$  actually provides an absolute maximum for  $J$ .

Of course, we are interested in the properties of  $y$  rather than  $J$ . We note that

1.  $\psi_0(t)$  satisfies the constraints, Equations [1]; in fact, the values of  $\lambda$  and  $\mu$  are chosen so that this is the case. (We shall assume the total impulse capability, *clnr*, of the rocket to be great enough, compared to  $gT$  and  $u_{x,0}$ , so that such values for  $\lambda$  and  $\mu$  exist.)
2.  $\psi_0(t)$  maximizes  $J$  relative to all other admissible functions, regardless of whether they satisfy the constraints.
3. It then follows that  $\psi_0(t)$  maximizes  $J$  relative to the more restricted class of functions which satisfy the constraints.
4. For functions which satisfy Equation [1], maximizing  $J$  is equivalent to maximizing  $y$ .

We conclude that  $\psi_0(t)$  maximizes the orbit altitude compared to all other thrust attitude programs which satisfy Equations [1].

<sup>a</sup>  $\lambda > 0$  simply means that  $\psi_0$  is a monotone decreasing function of  $t$ .

## Reliability Concepts in Rocket Power Controls Design

(Continued from page 640)

ATO or missile rocket system has given way to the sturdy, long-life component where rocket reliability is essential.

In the development of component reliability, functional endurance tests of all components used in "series" in the engine receive primary attention, and redundant subsystems are tested also as integral components. The basic functional endurance-test plan is designed to detect failures indicative of an insufficient safety margin. The environmental and operating overload test conditions adopted are determined on the basis of the actual safety and reliability margin considered necessary. The actual "overload" test conditions used in terms of force, pressure, vibration amplitude, extreme temperature, and timing are established by increasing the normal design-limit values to provide the desired reliability and safety margin. Failure is defined as the point at which out-of-tolerance operation begins.

Ten assemblies of each component and subsystem are set up first for endurance testing; all critical environment factors are simulated. Automatic recording of cycles and critical performance parameters is used to show initial deviation from specification performance. Each component is subjected to testing, modification or substitution, and retesting until it has achieved its specified reliability margin. Following this, sufficient additional samples incorporating the improvements are subjected to testing in order to establish statistical reliability. The component design is then released for incorporation in the engine reliability test program.

# Structural Materials for Missile Applications at Very High Temperatures

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Some short-time mechanical properties of copper, iron, molybdenum, tantalum and graphite were determined at temperatures up to the melting points of the metals and up to 5200 F for the graphite. These data are intended for use in the design of missile components that are exposed to aerodynamic heating. For periods of time up to 5 min, the load-carrying abilities of tantalum and molybdenum exceed those of iron and copper at equal temperatures. Although tantalum and molybdenum have appreciable strength in the temperature range 4000 to 5000 F, their service lives are limited because they are rapidly oxidized at these temperatures. Graphite has relatively low strength at room temperature, but it retains this strength with increasing temperatures up to 4400 F and higher. The graphite is oxidized at a much slower rate than the tantalum and molybdenum. In addition to mechanical and chemical properties, thermal properties must be considered in choosing materials for missile structures.

## Introduction

AS A result of aerodynamic heating, the temperatures of certain components in high speed missiles increase with increasing velocities. In some instances, these components must be designed to operate for short periods of time at the highest velocities and temperatures compatible with structural stability. The structural stability depends upon the mechanical properties under the operating conditions. The operating conditions, in addition to very high temperatures, include rapid heating, short times at temperature, and moderate to rapid rates of loading. Little or no information is available on the behavior of structural materials under these conditions of temperature, time and loading. Yet such knowledge is essential for the proper design and operation of high speed missiles.

The purpose of the work discussed in this paper was to determine the short-time tensile, creep and fracture properties of several structural materials at temperatures approaching their melting points. These properties determine, to a large extent, the suitability of the materials for applications in missile structures.

## Scope

The materials and types of tests and test conditions to which they were subjected are outlined in Table 1. All of the materials were tested in relatively pure form. Electrolytic-tough-pitch copper was used. The molybdenum had been made by the arc-vacuum casting method prior to rolling. It had a carbon content of 0.020 per cent. The tantalum was commercially pure and had been made by a powder metallurgical method before it was rolled. The ingot iron contained 0.018 per cent carbon plus very small percentages

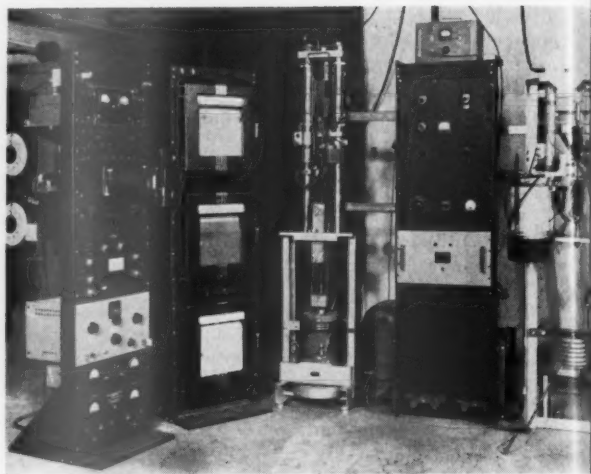


Fig. 1 Testing machine for tensile, creep and fracture tests

of manganese, phosphorus, sulfur, silicon and copper. In addition to the molded graphite, three types of extruded graphite were tested. The test results for only the molded graphite are reported since its mechanical properties were superior to all of the extruded types that were tested.

Since the test results were intended for use in applications involving aerodynamic heating, all tests were carried out in air atmospheres.

The tensile and creep-test specimens were heated to test temperature within 20 sec. The tensile specimens, which were held for periods of 10 and 90 sec at temperature, were then loaded at strain rates of 0.10 and 0.00005 in./in./sec. These strain rates caused the specimens to rupture in about 1 sec in the rapid tests and in about 30 min in the slower tests.

In the creep tests, the specimens were loaded at room temperature, heated to test temperature, and then held at test temperature until failure or for a maximum period of 5 min. Strain was measured continuously from the start of heating. In the fracture tests, the specimens were dead-weight loaded at room temperature and then heated continuously at three different rates to failure. No attempt was made to use the same three heating rates at each load but rather to heat at rates that would produce failures in times of 5, 20 and 90 sec from the start of heating. Strain and temperature were recorded continuously as functions of time during the fracture tests.

The dimensions of the gage sections of the test specimens were 2 in.  $\times$   $\frac{3}{4}$  in.  $\times$  0.064 in. for the sheet metals and 2 in.  $\times$  0.505 in. diam for the graphite.

## Test Equipment

Since no commercial testing apparatus was available that was capable of fulfilling the desired testing conditions, a unique high speed, high temperature testing apparatus was

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Table 1 Outline of test program

Materials <sup>1</sup>	Tests <sup>2</sup>	Test temp. range, °F	Test conditions
Copper, electrolytic tough pitch	Tensile, creep, fracture	75-1835	<i>Tensile</i>
Molybdenum, arc-vacuum cast		75-4500	Heating time, 20 sec
Tantalum, pressed and sintered		75-5000	Exposure times, <sup>3</sup> 10 and 90 sec Strain rates, 0.00005 and 0.1 in./in./sec
Ingot iron	Tensile, creep	75-2525	<i>Creep</i>
Molded graphite		75-5200	Heating time, 20 sec Max. test time, 5 min
			<i>Fracture</i> Heating rates to produce fracture in 5, 20 and 90 sec

<sup>1</sup> Copper and tantalum were rolled and annealed sheet, approximately 0.064-in. thick. Molybdenum and iron were cold-rolled sheet, approximately 0.064-in. thick.

<sup>2</sup> All tests carried out in air atmospheres.

<sup>3</sup> Refers to time at temperature before loading was started.

developed. This apparatus, which is described in more detail elsewhere,<sup>2</sup> is shown in Fig. 1. Briefly, the test equipment included a screw-driven tensile machine for the tensile tests and a dead-weight loading frame for the creep and fracture tests. The tensile machine was driven by a d-c motor through appropriate speed reducers. All specimens were resistance heated by means of current supplied through a welding transformer. Specimen temperature was measured by means of thermocouples, which were flash welded to the specimens, or by radiation pyrometers. Both the thermocouples and radiation pyrometers actuated a temperature recorder-controller that worked in conjunction with the resistance heating unit to control temperature at any desired level. In some tests at very high temperatures, an optical pyrometer was also used as an extra check on the specimen temperature.

In the tensile tests, load was transduced by a strain-gage

load cell. In all types of tests, strain was transduced by novel "spring-strain-gage" extensometers. The outputs from these transducers were amplified and fed into suitable electronic recorders that recorded stress-strain curves in the tensile tests and time-strain curves in the creep and fracture tests.

A typical tensile stress-strain curve is shown in Fig. 2. Curves of this type were obtained by feeding the calibrated signals from the stress and strain transducers into a cathode-ray tube and photographing the trace on a Land camera. In order to obtain a record of the strain rate, signals of calibrated frequency were applied to the Z-axis of the cathode-ray tube to produce dashes on the photographic record.

In order to provide some useful design data to the Air Force at the earliest possible moment, time was not taken to perfect the equipment for all of the desired test conditions. For example, the extensometer could be used to detect strain at the gage points of the specimens to a maximum testing temperature of only 2800 F. At higher testing temperatures strain was not measured on the metallic materials, and, for tests on graphite, strain was detected at the crosshead of the testing machine. On tantalum and molybdenum, some of the elevated-temperature tests that involved slow strain rates or long exposure times could not be carried out because the rapid oxidation of the metal either consumed the specimen before the test was completed or interfered excessively with temperature control. Fracture tests were not carried out on graphite because some of the required heating rates and fracture temperatures were beyond the capacity of the testing equipment. Subsequent work is being carried out to overcome these limitations in the equipment.

Repeated calibrations of the extensometers indicated that strain measurements taken at the gage points of the specimens were accurate within  $\pm 3$  per cent. The load cells, which were also calibrated frequently, were accurate within  $\pm 2$  per cent of the indicated values. The dead-weight loading frame was designed for an over-all loading accuracy of  $\pm 2$  per cent. Comparative tests, which were run on specimens with strain measurements taken both at the gage points and at the crosshead, provided a correction factor to apply to strain measurements taken at the crosshead. These comparative tests indicated that the accuracy of the strain measurements taken at the crosshead was probably no better than  $\pm 40$  per cent when the correction factor was applied.

Some inaccuracy was probably introduced into the total elongation values of the metal tensile specimens due to localized heating in the area that necked down after the ultimate load was reached. When this reduction in area occurred, the increased current density caused a localized increase in temperature, which undoubtedly changed the elongation characteristics in the area adjacent to the fracture. Since this

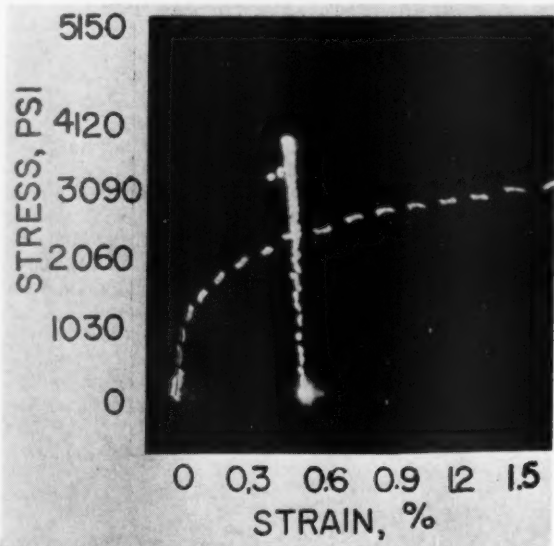


Fig. 2 Typical tensile stress-strain curve for ingot iron at 2450 F and at a strain rate of 0.1 in./in./sec; records of this type were obtained by photographing oscilloscope trace with a land camera



Fig. 3 Molybdenum specimen at 3000 F

phenomenon occurred after the ultimate load was reached, none of the other tensile properties were affected. Corresponding temperature nonuniformities that occurred near the end of the creep and fracture tests on the metal specimens probably had minor effects on the failing times in these tests. Because of their lack of ductility, no reduction of area occurred in the graphite specimens. Due to the small inherent inaccuracies in load and strain measurements and to individual variations in interpreting the elastic portion of the stress-strain curves, the maximum inaccuracy in the modulus of elasticity determinations was approximately  $\pm 10$  per cent.

Since the test specimens were resistance heated in open air, heat losses at the surface resulted in a temperature gradient from the center to the surface of the specimens. It was shown experimentally that at the highest testing temperatures this gradient was about 100 F in the copper and 250 F in the iron. It is estimated that the gradients were 400 F in the tantalum and 250 F in the molybdenum at the highest test temperatures. At lower temperatures, the

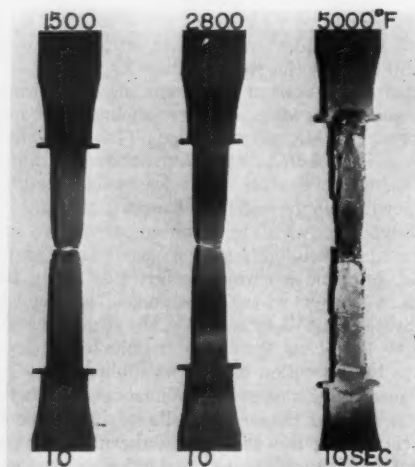


Fig. 4 Surface condition of tantalum specimens that were loaded rapidly to failure after 10-sec exposures to air at various temperatures

gradients were probably proportionately less. The magnitude of the temperature gradient in the graphite is not known, but it was probably considerably less than those in the metals because the graphite specimens were surrounded with a radiation shield. All reported test temperatures are those measured at the surface of the specimens.

## Test Results

### Oxidation

Molybdenum and tantalum oxidized rapidly at elevated temperatures and their usefulness is limited probably more by this phenomenon than by their mechanical properties. In both of these materials, the rate of oxidation, which was appreciable at temperatures between 1500 and 2000 F, increased with increasing temperatures. If the specimens were held at elevated temperatures for sufficient periods of time, the oxidation continued through the entire cross section. The molybdenum oxide formed a white smoke as shown in Fig. 3. The tantalum oxide formed a porous solid layer on the surface of the specimens. Fig. 4 shows three tantalum specimens that had been held at elevated temperatures for

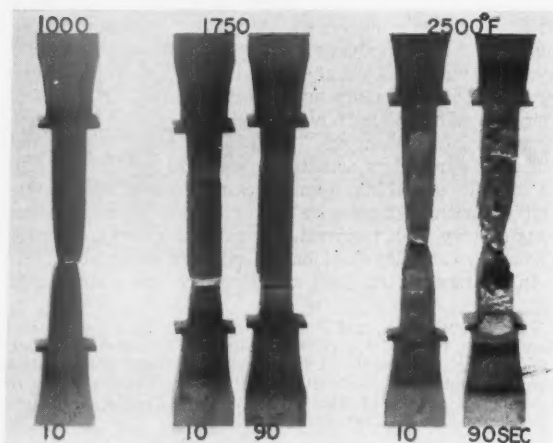


Fig. 5 Surface condition of ingot-iron specimens that were loaded rapidly to failure after various exposures to air at elevated temperatures

only 10 sec. At 1500 F no visible oxide had formed; at 2800 F a small amount of oxidation had occurred; at 5000 F the oxidation had badly damaged the metal.

Iron specimens also became severely oxidized at temperatures approaching their melting points. Fig. 5 shows that the oxidation was negligible at 1000 F, relatively minor for short periods of time at 1750 F, and quite severe at 2500 F. At temperatures near their melting points, the tantalum and molybdenum oxidized more rapidly than the iron.

Oxidation of copper and graphite specimens was relatively minor within the short time periods involved in the various tests. Fig. 6 shows that some oxidation of copper occurred as low as 800 F, but even at 1825 F only two or three thousandths inch of the cross section became oxidized. At 5200 F, only a small amount of graphite was oxidized as shown in Fig. 7.

### Tensile Properties

Figs. 8 and 9 show the ultimate tensile strength and 0.2 per cent-offset yield strength of the test materials at different temperatures, strain rates and exposure times. Graphite

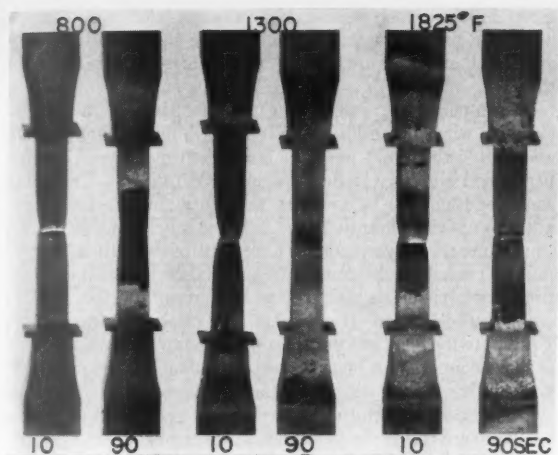


Fig. 6 Surface condition of electrolytic-tough-pitch copper specimens that were loaded rapidly to failure after various exposures to air at elevated temperatures

had no yield point since it underwent no plastic deformation when loaded to failure.

At the lower test temperatures, the strength of the test materials decreased in the following order: molybdenum, tantalum, iron, copper and graphite. At temperatures above 3200 F, the strength of the tantalum was slightly higher than that of the molybdenum. All of the metals decreased in strength continuously with increasing temperatures. The graphite, which had comparatively low strength at room temperature, maintained constant strength with increasing temperature up to 5200 F at the rapid strain rate and up to 4400 F at the slow strain rate. The strength properties of all of the metals were considerably higher at the rapid strain rate than at the slow strain rate. This effect of strain rate is common in metals. Except at temperatures above 4400 F, the effect of strain rate on the strength of graphite was less pronounced. The variation in exposure time from 10 to 90 sec at temperature had negligible effects on the tensile strength properties of the various test materials.

The modulus-of-elasticity values of the various materials



Fig. 7 Surface condition of specimens of molded and extruded graphite that were loaded rapidly to failure after 10-sec exposures to air at 5200 F

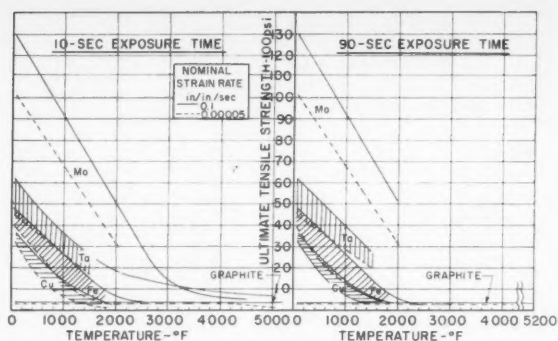


Fig. 8 Effect of temperature on the ultimate tensile strength of five materials at different strain rates and exposure times in air atmospheres; specimens were heated to test temperatures within 20 sec

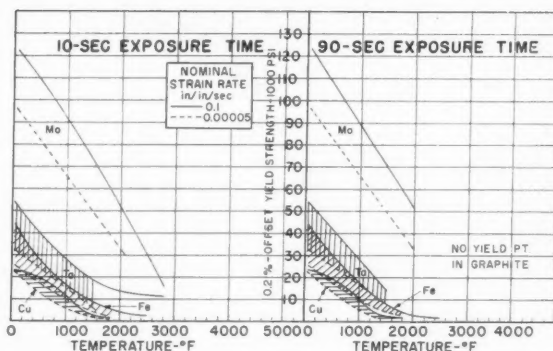


Fig. 9 Effect of temperature on the 0.2% offset yield strength of four materials at different strain rates and exposure times in air atmospheres; specimens were heated to test temperature within 20 sec

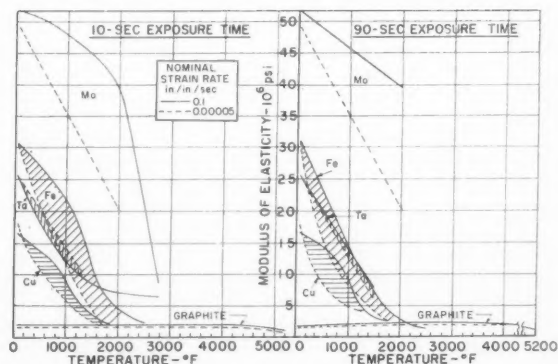


Fig. 10 Effect of temperature on the tensile modulus of elasticity of five materials at different strain rates and exposure times in air atmospheres; specimens were heated to test temperature within 20 sec

under different test conditions are shown in Fig. 10. With the exception of the reversal in positions of iron and tantalum at the lower temperatures, the relative order was similar to that of the tensile-strength properties. As in the case of strength properties, the modulus values of the metals decreased continuously with increasing temperatures, and the modulus of the graphite remained constant or increased

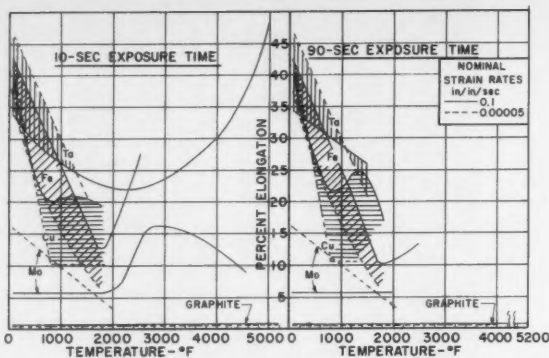


Fig. 11 Effect of temperature on the per cent elongation of five materials at different strain rates and exposure times in air atmospheres; specimens were heated to test temperature within 20 sec

slightly with increasing temperatures up to about 4400 F. Between 4400 and 5200 F, the modulus of the graphite decreased slightly.

At room temperature, the variation in strain rate had little or no effect on modulus-of-elasticity values. At elevated temperatures, the modulus of molybdenum, iron and copper increased with increasing strain rate. This variation in modulus with different strain rates is probably not a real variation in elastic properties. It is likely that, at the slow strain rate and high testing temperatures, some creep deformation was added to the elastic deformation. Such additional strain during the elastic portion of the tests causes an apparent decrease in the modulus values. In tests at the rapid strain rates, little or no creep deformation occurred during elastic straining since this portion of the tests lasted for only a fraction of a second. The moduli of tantalum and graphite were not greatly affected by variations in strain rate. Variations in exposure time had little significant effect on modulus values.

The total permanent elongations obtained on the tensile test specimens are shown in Fig. 11. At room temperature, the ductility of tantalum, copper and iron were about equal and considerably higher than that of molybdenum. The graphite had no plastic elongation at any of the test temperatures. With increasing temperatures, the tantalum and iron decreased in elongation and then increased as shown in Fig. 11. The elongation of the copper decreased as temperature was increased to 800 F and remained constant with further increases in temperature. The increase in ductility of molybdenum as temperature was increased from 2000 to 2800 F was, undoubtedly, a result of the fact that the cold-rolled structure was annealed in that temperature range.

The variation in strain rate from 0.00005 to 0.1 in./in./sec did not have consistent effects on ductility. In general, the elongation of the iron and copper was higher at the higher strain rate. At low temperatures, the tantalum and molybdenum had higher elongation at the lower strain rate, but at temperatures between 1300 and 1600 F the ductility of these metals at the slow strain rate fell below that obtained at the rapid strain rate.

The variation in exposure time from 10 sec to 90 sec at the test temperature had very little effect on per cent elongation in all of the test materials.

#### Creep Properties

Some comparisons of the short-time creep-rupture properties of the test materials at several loads and temperatures are shown in Fig. 12. In this graph, the heights of the vertical bars represent the rupture times of various test materials at different temperatures and at tensile loads of 200, 2000 and 10,000 psi. The bars are divided into sections to represent

three different creep deformation ranges as shown on Fig. 12. Since the maximum testing time was 5 min, rupture times greater than 5 min are represented by arrows at the tops of the bars.

At temperatures approaching their melting points, copper and iron withstood a load of 200 psi for 5 min without failure and with less than 1 per cent creep deformation. Graphite performed similarly under this load at 5200 F. Under the same load at 4500 F, however, the tantalum failed in 26 sec and the molybdenum in 48 sec. These quick ruptures of the tantalum and molybdenum at temperatures near their melting points probably do not represent inherent creep-rupture properties, since the failures were a result of the wasting away by oxidation of the 0.064-in.-thick test specimens. Undoubtedly, the rupture times would have been much longer if the specimens had been protected against oxidation. Also, the relative rate at which the metal is wasted away decreases with decreasing surface area-to-volume ratio. At the same temperature and stress, thicker specimens, for example, would not have failed so quickly. Since graphite and iron are oxidized at slower rates than tantalum and molybdenum, similar factors would probably have a less marked effect on their creep-rupture properties.

Under a load of 2000 psi, copper and iron specimens failed in less than 30 sec at temperatures near their melting points. When the temperatures was lowered to 1375 F in the copper and to 1900 F in the iron, neither material ruptured within 5 min. At the same load, graphite, which failed in 51 sec at 5200 F, did not fail within 5 min at 4750 F.

When tantalum and molybdenum were creep tested at 2500 F under a 10,000-psi load, both materials failed within 30 sec. At the same load, molybdenum elongated 5 per cent and failed in 2.5 min at 2150 F, and tantalum elongated 10 per cent but did not fail within 5 min at 2125 F. At lower temperatures, neither tantalum nor molybdenum crept as much as 1 per cent or failed within 5 min.

The results of these creep tests show that the short-time

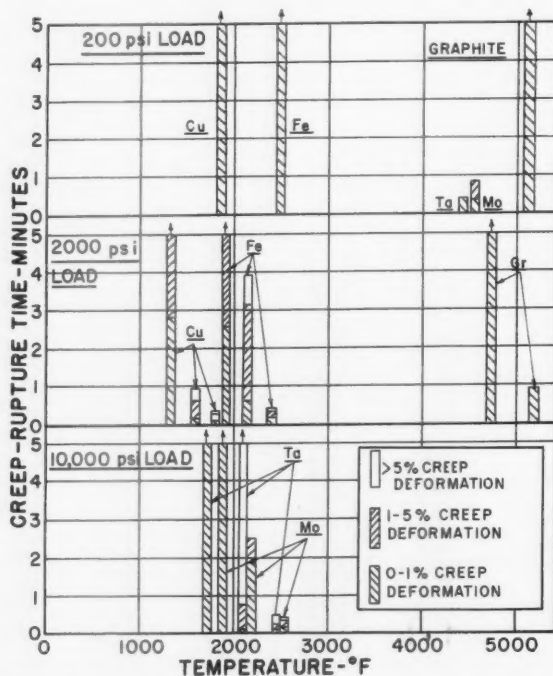


Fig. 12 Effect of temperature on creep-rupture time of five materials at various loads in air atmospheres; specimens were dead-weight loaded at room temperature and heated to test temperature within 20 sec

creep-rupture properties of tantalum and molybdenum are superior to those of iron and copper at the same temperatures. The molded graphite showed remarkable creep strength in that at 4750 F it supported a load equivalent to two-thirds its room temperature tensile strength for more than 5 min.

In order to provide more design data on copper, extra short-time creep tests were run at various stresses at temperatures of 1000, 1400, and 1800 F. The results are plotted in Fig. 13, which shows the rupture times and times required for various creep deformations at different stresses and temperatures. This plot shows that for service periods of 5 min or less the maximum safe loads for copper are between 7000 and 8000 psi at 1000 F, between 3000 and 4000 psi at 1400 F and between 1000 and 2000 psi at 1800 F. These maximum safe loads are approximately equivalent to the ultimate tensile strength of the copper at the same temperatures at the slow strain rate. The ultimate tensile strengths at these temperatures at both slow and rapid strain rates are shown on the left-hand side of Fig. 13.

### Fracture Tests

The results of fracture tests on the test metals are shown in Fig. 14. In this plot, the temperatures at which the fracture-test specimens failed are shown as functions of stress. For each stress level, three failing temperatures are shown, which were obtained by heating at three different rates to produce fractures in times of 90, 20 and 5 sec. The lower the stress, the higher the temperature to which the specimens could be heated before they ruptured. In all of the metals at all stress levels, the failing temperature increased with increasing heating rate. Also shown in Fig. 14 are the ultimate tensile strengths of the test metals at different temperatures and at two strain rates. The relationships between fracture temperature and stress correspond quite closely to the relationships between temperature and values of ultimate tensile strength that were obtained at a strain rate of 0.1 in./in./sec.

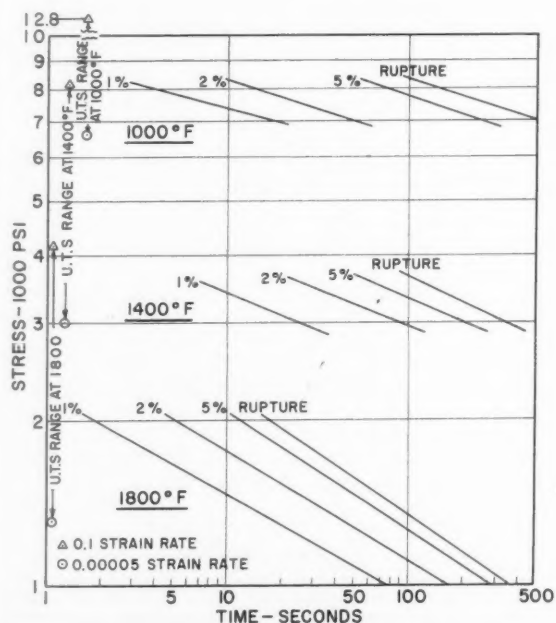


Fig. 13 Short-time creep and rupture properties of electrolytic-tough-pitch copper showing time for various amounts of creep at various stresses and temperatures in air atmospheres; specimens were loaded at room temperature, heated to test temperature within 20 sec, and then creep measurements were made for a maximum of five minutes; ultimate tensile strengths shown for comparison

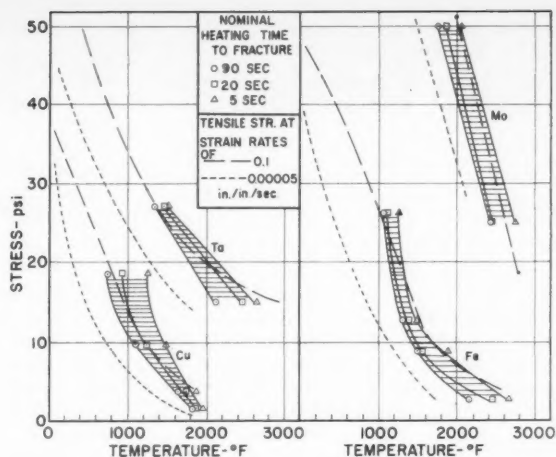


Fig. 14 Fracture temperatures of four metals dead-weight stressed at room temperature and continuously heated to fracture in 5, 20 and 90 sec; tensile strengths also shown for comparison

### Conclusions

The information contained in this report is intended as an aid for designers in the selection of materials for and design of components for missile structures. Definite conclusions regarding the best materials for missile structures cannot be made on the basis of this investigation alone. Thermal properties in addition to mechanical and chemical properties are of prime importance in making a selection of this type.

The selection of materials must be based upon a correlation between the mechanical, chemical and thermal characteristics of the structural materials and the conditions of temperature, time and loading under which the missile will operate. None of the materials tested have optimum properties for a wide range of operating conditions. Tantalum and molybdenum have appreciable strength at temperatures between 4000 and 5000 F, but their service lives in air at these temperatures are quite limited because they are oxidized very rapidly. For periods of a few seconds in this temperature range, the tantalum and molybdenum sheet had more strength than the graphite bar. The graphite, which oxidized more slowly, was superior for load-supporting applications of more than 1 min in air at these temperatures. Graphite has very low strength at room temperature but is unusual in that it retains this strength at temperatures up to 4400 F and higher. The brittleness of graphite under all conditions, however, might mitigate its advantages of stability and strength at very high temperatures.

At equal temperatures, the load-carrying abilities of molybdenum and tantalum considerably exceed those of iron and copper for periods up to 5 min. However, the excellent thermal properties of copper, which are reported elsewhere, are probably a distinct advantage for this material.

Of equal importance to knowledge of the characteristics of missile structural materials under operating conditions is the development of improved materials for these applications. If suitable protective coatings are developed for tantalum and molybdenum, these metals will be promising for applications at temperatures up to the range 4000 to 5000 F.

### Acknowledgment

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# Some Considerations of Film Cooling for Rocket Motors<sup>1</sup>

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It is demonstrated experimentally that increasing the film-cooled length of a cylindrical section requires more than a proportional increase in the film-coolant flow rate. It is shown that if the heat transfer rate to the liquid film is known, an ideal film-coolant flow rate may readily be calculated, which is proportional to the average film-cooled length. The actual film-coolant flow rate may then be determined from the graphs presented in the paper. Circumferential variations in the length of the liquid film are taken into account by means of the *profile effectiveness* which depends upon the design of the film-coolant injector and the main propellant injector. A method for estimating the heat transfer to the film coolant is presented. Experimental data are also presented for the loss in specific impulse when water is used as the film-coolant. The rocket motor experiments reported herein were conducted with a 500-lb-thrust, 500-psia combustion pressure rocket motor using white fuming nitric acid and jet engine fuel (JP-4) as the propellants.

## Nomenclature

- $A_a$  = actual film-cooled area
- $c_p$  = specific heat at constant pressure of liquid film coolant
- $D$  = diameter of combustion chamber
- $d$  = thickness of liquid film
- $d^+$  = dimensionless thickness of liquid film
- $F$  = thrust
- $f$  = Fanning friction coefficient
- $g$  = acceleration of gravity
- $h_f$  = heat transfer coefficient for film cooling
- $\Delta h_e$  = specific heat of vaporization of coolant corresponding to combustion pressure
- $I$  = specific impulse as defined by Equation [15]
- $\delta I$  = per cent loss in specific impulse due to film cooling
- $I_0$  = "effective specific impulse" for zero film-coolant flow
- $I_{FC}$  = specific impulse for any particular value of film-coolant flow
- $L$  = film-cooled length
- $\bar{L}$  = average film-cooled length
- $Pr$  = Prandtl number of combustion gases based on bulk conditions
- $q$  = heat flux, or heat transferred per unit area
- $Q_1$  = heat which the coolant can absorb as a liquid
- $Q_2$  = heat transferred from the combustion gases
- $R$  = ratio of the maximum gas velocity to the gas velocity at the border of laminar sublayer and turbulent core
- $R'$  = ratio of mean gas velocity to gas velocity at border of laminar sublayer and turbulent core
- $St$  = Stanton number of combustion gases based on bulk conditions
- $t_b$  = temperature of combustion gases evaluated at bulk conditions
- $t_s$  = saturation temperature of liquid film coolant corresponding to the combustion pressure

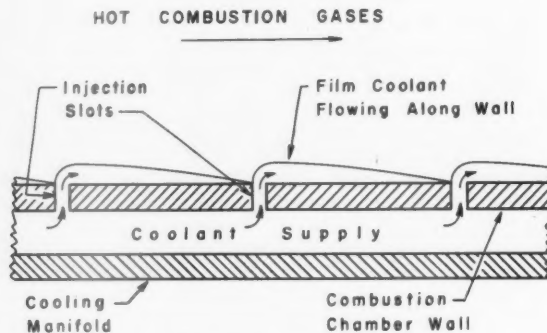


Fig. 1 Film-cooled combustion chamber

- $t_1$  = temperature of coolant at supply conditions
- $u$  = velocity at any distance  $y$  as noted in text
- $u^+$  = dimensionless velocity defined by Equation [7]
- $\dot{W}$  = actual film-coolant flow rate
- $\dot{W}'$  = ideal film-coolant flow rate
- $\dot{W}^+$  = dimensionless film-coolant flow rate
- $\dot{W}_o$  = oxidizer flow rate
- $\dot{W}_f$  = fuel flow rate
- $y$  = radial distance measured as noted in text
- $y^+$  = dimensionless wall distance as defined by Equation [8]
- $\delta$  = thickness of laminar sublayer
- $\epsilon_p$  = profile effectiveness as defined by Equation [4]
- $\epsilon_s$  = stability effectiveness as defined by Equation [5]
- $\mu$  = dynamic viscosity of liquid film coolant
- $\rho$  = density of the liquid film coolant
- $\tau$  = shear stress evaluated at any distance  $y$  as noted in text
- $\tau_0$  = shear stress evaluated at  $y = 0$

## 1 Introduction

RECENT trends in rocket engine development are in the direction of utilizing higher combustion pressures and/or higher energy propellants (5, 22).<sup>4</sup> Operation under those conditions, however, aggravates the problem of cooling the rocket motor. It appears that with regenerative cooling the maximum usable specific impulse obtainable from a propellant combination will be limited by the ability of the particular propellant employed as the liquid coolant to absorb the required heat flux (1, 2, 3, 4).

## 2 Film Cooling

It has been pointed out in the literature that the limitations of regenerative cooling may be overcome by film cooling with a liquid coolant (22, 23).

Fig. 1 illustrates a section through the wall of a cylindrical combustion chamber where the hot combustion gases flow past the interior surface and a liquid film coolant is supplied through slots from a manifold to cover that surface. The

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<sup>2</sup> This work was sponsored by the Office of Naval Research, Power Branch, Department of the Navy, under Contract N7 onr 39418. Reproduction in full or in part is permitted for any use of the United States Government.

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film coolant should be so injected that after entering the combustion chamber it will flow along the surface and be evaporated gradually as it flows downstream. Before complete evaporation occurs, a new supply of liquid film coolant is introduced. As long as there is liquid between the hot gas and the surface, the temperature of the interior wall will not exceed the boiling point of the liquid.

The studies of film cooling reported herein may be divided into two parts: (a) The determination of the film-coolant flow rates required for cooling a given length of a cylindrical section using water as the film coolant,<sup>5</sup> and (b) the determination of the decrease in the specific impulse obtained from a rocket motor due to film cooling a portion of its combustion chamber.

The heat which the liquid coolant can absorb, denoted by  $\dot{Q}_1$ , is given by

$$\dot{Q}_1 = \dot{W}'[c_p(t_s - t_i) + \Delta h_v] \dots \dots \dots [1]$$

On the other hand, the heat which the coolant must absorb from the hot gases flowing past it, is given by<sup>6</sup>

$$\dot{Q}_2 = \pi D \bar{L} h_f (t_b - t_s) \dots \dots \dots [2]$$

Equating Equations [1 and 2] and solving for the ideal film-coolant flow rate  $\dot{W}'$  yields

$$\dot{W}' = \frac{\pi D \bar{L} h_f (t_b - t_s)}{c_p(t_s - t_i) + \Delta h_v} \dots \dots \dots [3]$$

Equation [3] cannot be applied directly in practice, as will be seen from the considerations which are discussed in the following paragraphs.

**Profile effectiveness:** Equation [3] includes the average

film-cooled length  $\bar{L}$ . Reference to Fig. 2, which is a pattern view of a film-cooled cylindrical duct, shows that in general the length of the liquid film may vary from point to point around the circumference of the combustion chamber. The minimum length of cylindrical duct which is film cooled is denoted by  $L$ , and will be termed the "film-cooled length." From Fig. 2 it is apparent that the average film-cooled length  $\bar{L}$  is equal to the total film-cooled area  $A_a$ , divided by the circumference of the duct  $\pi D$ . It is convenient to introduce the profile effectiveness  $\epsilon_p$ , which measures variations in the length of the liquid film. By definition

$$\epsilon_p = L/\bar{L} \dots \dots \dots [4]$$

The magnitude of the profile effectiveness depends upon (a) the characteristics of the film-coolant injector, (b) the condition of the surface to be cooled and (c) the characteristics of the injector employed for introducing the main propellants into the combustion chamber of the rocket motor.

If the film-coolant injector provides a uniform film distribution, and if the surface to be film cooled is relatively smooth, then  $\epsilon_p$  depends only upon the characteristics of the main propellant injector. The latter influences the value of  $\epsilon_p$  if it causes hot spots to be formed because of nonuniformities in the combustion process.

**Stability effectiveness:** A second consideration which must be taken into account arises from the phenomena associated with film instability. Reference to Equation [3] shows that for a given liquid film coolant, the denominator of the right-hand side is constant. For a given rocket motor, burning a given set of propellants at a fixed mixture ratio and combustion pressure, the numerator, except for  $\bar{L}$ , is also constant. It would, therefore, be inferred that the average film-cooled length  $\bar{L}$  would vary linearly with the film-coolant flow rate. Experiments show, however, that an increase in the film-cooled length requires more than a proportionate increase in the film-coolant flow rate.<sup>7</sup> The nonlinearity may be attributed primarily to one of two factors.

First, as the film-coolant flow rate is increased, there may occur separation of the liquid film coolant from the surface close to the location where it is injected. In that event a large portion of the liquid film coolant penetrates into the main gas stream and the quantity of liquid actually required for film cooling the surface is increased. For reasonable film-coolant flow rates, experiments indicate that the aforementioned condition is unlikely to occur (13, 27).

It is far more probable that the nonlinearity between film-cooled length and film-coolant flow rate arises from the formation of waves and ripples on the free surface of the liquid film. At low values of film-coolant flow rate the liquid flow is laminar and the surface of the film is relatively smooth. As the flow rate is increased the flow becomes turbulent and waves or ripples begin traveling along the surface of the film. Further increase in the film-coolant flow rate intensifies the wave motion, with the consequence that portions of the waves are "torn away" by the high velocity gases flowing past them; that phenomena will be termed "droplet break-away." Such waves (or ripples) increase the friction coefficient between the afore-mentioned gases and the liquid film (8) and also increase the effective area for transferring heat from the hot gases to the film.

The factors discussed here are effective in causing the required film-coolant flow rate to be larger in the actual case than in the ideal case. It is convenient, therefore, to introduce the stability effectiveness  $\epsilon_s$ , which is defined by

$$\epsilon_s = \dot{W}'/\dot{W} \dots \dots \dots [5]$$

where  $\dot{W}$  is the film-coolant flow rate required in the actual case.

<sup>7</sup> This statement is confirmed later with experimental evidence (see Figs. 3 and 7).

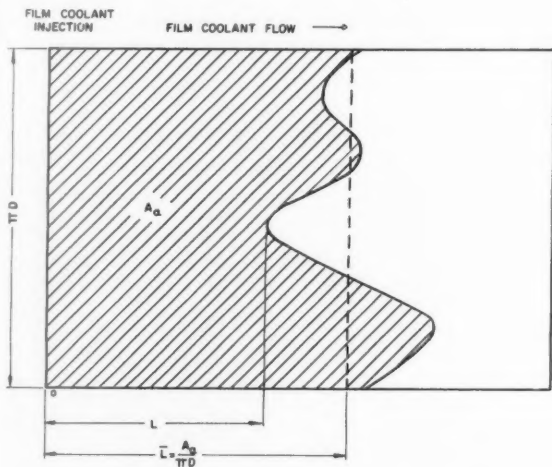


Fig. 2 Pattern view of film-cooled test section

<sup>5</sup> Experiments have been conducted at Purdue University with JP-4, WFNA and liquid ammonia as film coolants but the results are not reported here.

<sup>6</sup> It is assumed that all of the heat transferred from the combustion gases is transferred to the liquid coolant and that none is transferred through the walls under equilibrium conditions. It can be shown that, in general, less than approximately 3 per cent of the total heat transferred from the combustion gases will be conducted through the coolant and the wall because of the extremely high resistance of the still air film on the outside of the wall.

Substituting Equations [4, 5] into Equation [3] to eliminate  $\bar{L}$  and  $W'$ , and solving for  $\bar{W}$ , gives

$$\bar{W} = \frac{\pi D L h_f (t_b - t_s)}{\epsilon_p \epsilon_s [c_p (t_s - t_i) + \Delta h_v]} \dots\dots\dots [6]$$

The only quantities that would normally be unknown in Equation [6] are the heat transfer coefficient  $h_f$ , the profile effectiveness  $\epsilon_p$ , and the stability effectiveness  $\epsilon_s$ . The next three sections present methods for determining  $h_f$ ,  $\epsilon_p$ , and  $\epsilon_s$ .

**3 Determination of Stability Effectiveness  $\epsilon_s$**

It has been shown that the stability effectiveness  $\epsilon_s$  depends upon the intensity of waves (or ripples) formed on the free surface of the liquid film. For a given liquid film coolant, it may be expected that the intensity of those ripples will depend upon the thickness of the film; the latter is related to the degree of turbulence in the film-coolant flow (6, 12, 15, 19, 28). Consequently, the stability effectiveness  $\epsilon_s$  may be expected to be a function of the thickness of the liquid film. If it is assumed that the flow in the liquid film can be characterized by some sort of dimensionless velocity distribution similar to that applicable for single phase flow,<sup>8</sup> then the stability effectiveness  $\epsilon_s$  should be dependent upon the dimensionless film thickness.<sup>9</sup> It can be shown that the dimensionless film thickness  $d^+$  depends upon the dimensionless film-coolant flow rate  $\bar{W}^+$ ; the equation for the latter is derived below (19). The dimensionless velocity  $u^+$  and the dimensionless distance  $y^+$  are defined by

$$u^+ = \frac{u}{\sqrt{\tau_0/\rho}} \dots\dots\dots [7]$$

and

$$y^+ = \frac{y}{\mu/\rho} \sqrt{\tau_0/\rho} \dots\dots\dots [8]$$

where  $u$  is the velocity in the liquid film at any distance  $y$  from the wall,  $\mu$  and  $\rho$  are the viscosity and density, respectively, of the film coolant, and  $\tau_0$  is the shear stress between the liquid film and the wall.

The weight rate of film-coolant flow  $d\bar{W}$  through an elementary area  $\pi D \, dy$  of the tube is given by

$$d\bar{W} = \rho u \pi D \, dy \dots\dots\dots [9]$$

Substituting for  $u$  from Equation [7] into Equation [9] and substituting for  $dy$  from the differentiated form of [8] gives

$$\frac{\bar{W}}{\pi D \mu g} = \int_{y^+=0}^{y^+=d^+} u^+ \, dy^+ = \bar{W}^+ \dots\dots\dots [10]$$

By definition, the right-hand side of [10] is the dimensionless film-coolant flow rate  $\bar{W}^+$ . It is apparent from [10] that  $\bar{W}^+$  is a function of the upper limit of integration  $y^+ = d^+$ . Since  $\epsilon_s$  is also a function of  $d^+$ , which is the dimensionless film thickness, it follows that

$$\epsilon_s = f(\bar{W}^+) = f\left(\frac{\bar{W}}{\pi D \mu g}\right) \dots\dots\dots [11]$$

The functional relationship given by Equation [11] must be determined experimentally. Consider the data presented

<sup>8</sup> It is not assumed in this paper that the dimensionless velocity distribution is the same as that for single phase flow, as reported by Nikuradse (18). It is only assumed that a dimensionless velocity distribution exists for the liquid film.  
<sup>9</sup> Reference (6) suggests that stable and unstable films might be related by the value of the dimensionless film thickness.

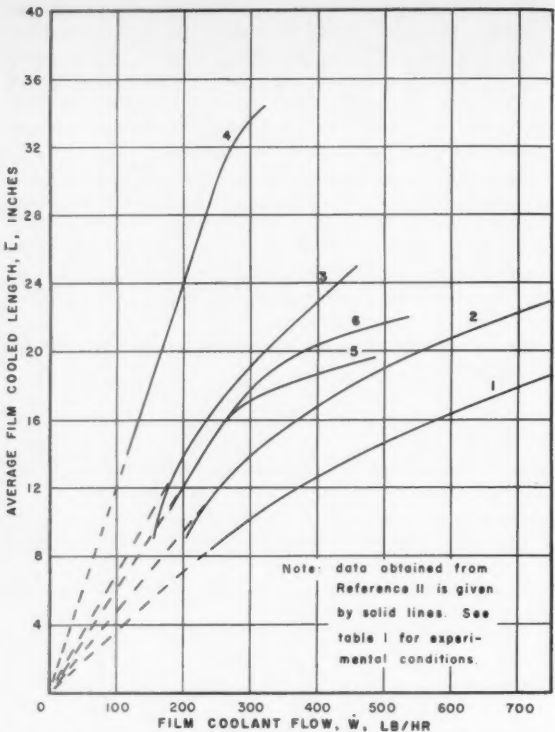


Fig 3 Average film-cooled length vs. coolant flow

in Reference (11) which were obtained from experiments with heated air flowing through a circular film-cooled duct. The experiments were conducted with water and ethylene glycol as the film coolants. Table 1 summarizes the experimental conditions.

Table 1				
Type of surface <sup>1</sup>	Curve	Coolant	Temperature, °F	Gas mass velocity, lb/hr ft <sup>2</sup>
Rough surface duct	1	water	1400	2.07 × 10 <sup>5</sup>
	2	water	1400	1.63–1.73 × 10 <sup>5</sup>
	3	water	800	2.67 × 10 <sup>5</sup>
	4	water	800	1.65 × 10 <sup>5</sup>
Roughened surface duct	5	water	800	2.63 × 10 <sup>5</sup>
	6	ethylene glycol	700	1.90 × 10 <sup>5</sup>

<sup>1</sup> The surface of the duct listed as rough was relatively smooth, but slightly wavy. The surface listed as roughened was the same duct after it had been etched by flowing ethylene glycol through the duct as a film coolant.

Fig. 3 presents the curves of  $\bar{L}$  as a function of  $\bar{W}$  reported by (11); the conditions corresponding to the numbers attached to the curves are listed in Table 1.

Reference to Fig. 3 shows that the  $\bar{L}$  is not a linear function of  $\bar{W}$ , except possibly when extrapolated to low values of  $\bar{W}$ . The nonlinearity is attributed to the stability effectiveness  $\epsilon_s$ . Eliminating  $\bar{W}'$  in Equation [3] by means of [5] yields

$$\bar{W} = \frac{h_f}{\epsilon_s} \cdot \left[ \frac{\pi D \bar{L} (t_b - t_s)}{c_p (t_s - t_i) + \Delta h_v} \right] \dots\dots\dots [12]$$

The curves of Fig. 3 can be utilized for evaluating the bracketed

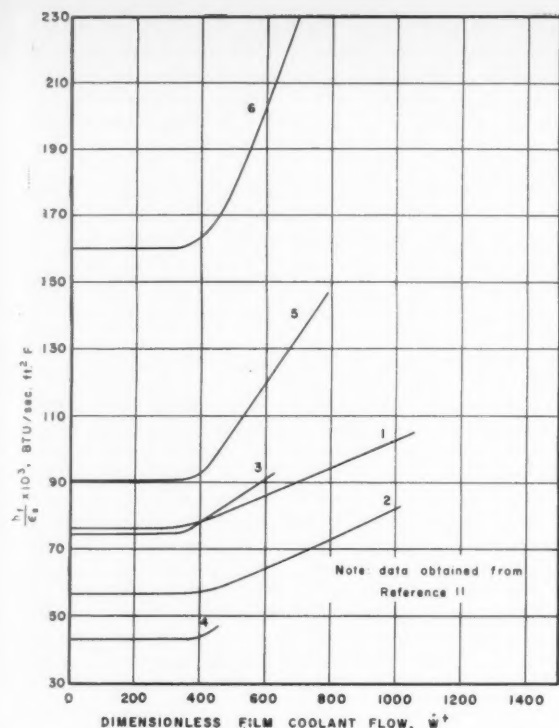


Fig. 4 Parameter  $h_f/\epsilon_s$  vs. dimensionless film-coolant flow rate

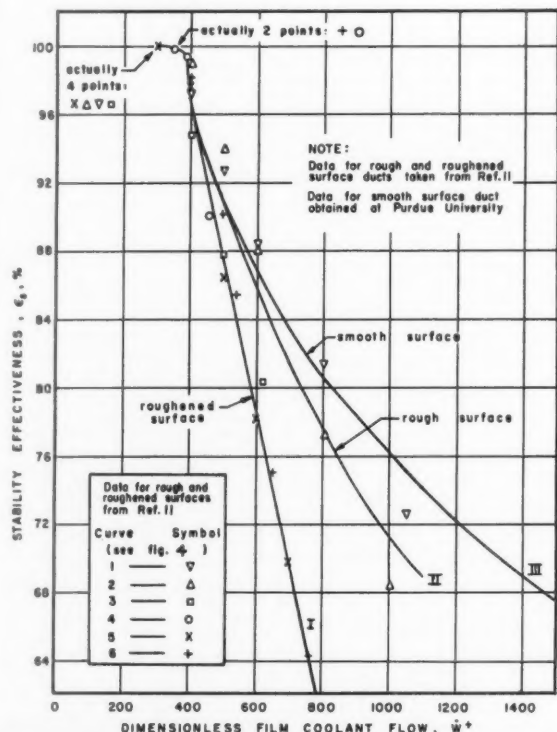


Fig. 5 Stability effectiveness vs. dimensionless film-coolant flow

eted expression in [12], and those curves can then be transformed into curves presenting

$$\dot{W}^+ = f\left(\frac{h_f}{\epsilon_s}\right) \dots \dots \dots [13]$$

Fig. 4 presents the corresponding curves of  $h_f/\epsilon_s$  as a function of  $\dot{W}^+$ , obtained from Fig. 3. It is seen from Fig. 4 that at low values of  $\dot{W}^+$  each curve is parallel to the  $\dot{W}^+$  axis. It may be assumed, therefore, that the portion of each curve that is parallel to the  $\dot{W}^+$  axis corresponds to the situation where the surface of the liquid film was essentially free of waves, so that the stability effectiveness  $\epsilon_s$  was unity. Consequently, the value of the parameter  $h_f/\epsilon_s$  for the aforementioned portion of each curve presented in Fig. 4 gives the heat transfer coefficient  $h_f$  for that particular curve.

From the values of  $h_f$  obtained from each curve, and assuming that  $h_f$  is independent of the film-coolant flow rate, the values of  $1/\epsilon_s$  as a function of  $\dot{W}^+$  are readily computed for each curve by dividing the ordinates by their corresponding value of  $h_f$ .

Fig. 5 presents the values of  $\epsilon_s$  as a function of  $\dot{W}^+$  calculated in the afore-mentioned manner. Curves I and II in Fig. 5 apply to the experiments listed in Table 1. Curve III was developed from data obtained at the Jet Propulsion

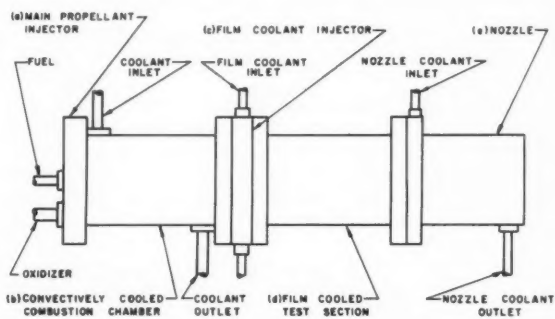


Fig. 6 Arrangement of elements of film-cooled rocket motor

Center, Purdue University, with the 500-lb-thrust rocket motor illustrated in Fig. 6. The experiments with the rocket motor were conducted at 500-psia combustion pressure using JP-4 fuel and white fuming nitric acid as the propellants; the film coolant was water.

Fig. 7 presents the film-cooled length as a function of the film-coolant flow rate  $\dot{W}$  for the same film-cooled rocket motor operated with two different film-coolant injectors; the injectors are discussed in a later section. The film-cooled length was determined from thermocouple measurements of the inside wall temperatures at the downstream end of a film-cooled test section. In an experiment the film-coolant flow rate was decreased in steps and those temperatures were noted. When the surface was completely covered with a liquid film the afore-mentioned thermocouples indicated temperatures corresponding to the boiling point of the liquid film coolant. When one or more of the thermocouples indicated a surface temperature which was rising above the boiling point of the film coolant, the experiment was terminated. Experiments were conducted with film-cooled test sections having different lengths and with different liquid film coolants. In all of the experiments the surface of the film-cooled test section was smooth.

The curves of Fig. 5, being generalized curves, are independent of the diameter of the tube (cylindrical combustion

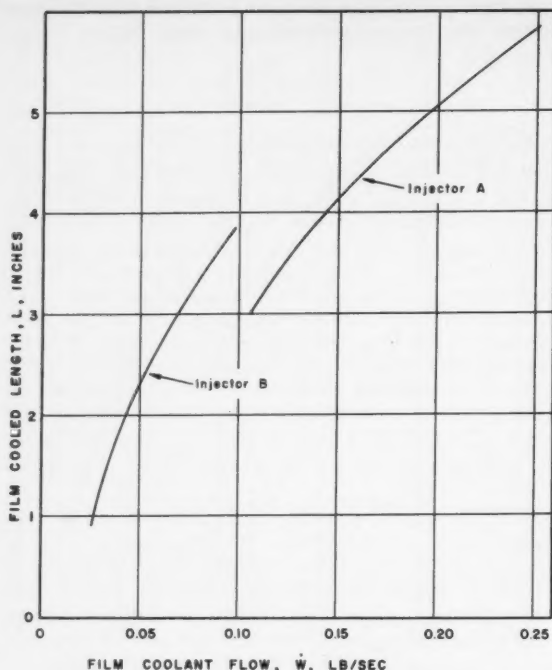


Fig. 7 Film-cooled length vs. film-coolant flow

chamber).<sup>10</sup> Fig. 5 also shows that the condition of the film-cooled surface has a relatively small effect upon the stability effectiveness. The curves indicate that the optimum operating regime, from the point of view of minimum film-coolant flow rate, is where  $\dot{W}^+ < 400$ . With  $\dot{W}^+ > 400$ , the stability effectiveness  $\epsilon_s$  decreases rapidly.

#### 4 Determination of Heat Transfer Coefficient $h_f$

The method employed for determining the heat transfer coefficient and its correlation is due to J. P. Sellers,<sup>11</sup> and the derivations are presented in (28). The following assumptions were made:

- 1 The heat transferred through the wall wetted by the liquid film is so small that it may be neglected.
- 2 Heat is transferred through the laminar sublayer by conduction alone.
- 3 The ratio of the shear stress  $\tau$  to the heat flux  $q$  at any distance  $y$  above the surface of the liquid film is constant.
- 4 At  $y = 0$ , or at the free surface of the liquid film, the shear stress is given by the conventional Fanning friction equation based on the main gas stream conditions.
- 5 The film-coolant flow rate is smaller than approximately 10 per cent of the hot gas flow.

It can be shown that the resulting heat transfer correlation is given by

$$St = \frac{(f/2)R'}{Pr + R - 1} \dots \dots \dots [14]$$

where  $St$  and  $Pr$  are the Stanton and Prandtl numbers for the

<sup>10</sup> It was shown (Ref. 10), where experiments were conducted with both 2- and 4-in. hot air ducts, that the data for both diameter ducts could be correlated under a given set of conditions as  $\bar{L} = f(\dot{W}/\pi D)$ . Therefore, it is apparent that the curves of Fig. 5 should be applicable to any diameter combustion chamber which is to be film cooled.

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hot gas, respectively,  $f$  is the Fanning friction coefficient at the interface of the liquid film and the combustion gases,  $R$  is the ratio of the maximum velocity of the hot gas to its velocity at the interface of the laminar sublayer and the turbulent core, and  $R'$  is the ratio of the mean velocity of the hot gas to its velocity at the interface of the laminar sublayer and the turbulent core.

The data obtained at low values of film-coolant flow (where  $\epsilon_s = 1.0$ ) by (24), and also those obtained at Purdue University, are correlated satisfactorily by Equation [14] for the values  $(R' \cdot f/2) = 0.0093$  and  $R = 4.50$ . It is worth noting, in passing, that for  $Pr = 1.00$  and highly turbulent flow ( $R \approx R'$ ), Equation [14] reduces the Reynolds analogy.

#### 5 Determination of Profile Effectiveness $\epsilon_p$

It was stated earlier that the profile effectiveness depends upon (a) the film-coolant injector, (b) the condition of the surface to be cooled and (c) the main propellant injector (hot spots).

Referring to Fig. 7, it will be noted that to cool a given length of test section, Injector A required a larger film-coolant flow rate than did Injector B. In other words, Injector B gave larger values of profile effectiveness.

Fig. 8 illustrates the design of Injector A. The liquid coolant flows through the hole in the top of the flange into the film-coolant manifold, and then flows through 72 milled grooves onto the "injector lip," where it is guided to flow in the same direction as the combustion gases. The film coolant then enters the cylindrical test section through the circumferential slot formed between the injector lip and the inner wall of the film-cooled test section. The experiments showed that the injector lip could not be cooled adequately and consequently deformed nonuniformly. The deformation varied from point to point around the circumference of the lip, with the result that there was a circumferential variation in the width of the slot through which the liquid was discharged into the cylindrical test section. Consequently, the liquid film did not extend the same distance from the discharge plane of the film-coolant injector at all points around the circumference of the film-cooled test section.

Fig. 9 illustrates the construction of Injector B; the latter was designed to overcome the disadvantages of Injector A. The liquid film coolant, after entering the film-coolant manifold, is injected into the combustion chamber through several tangential grooves, as shown in Section A-A; an enlarged

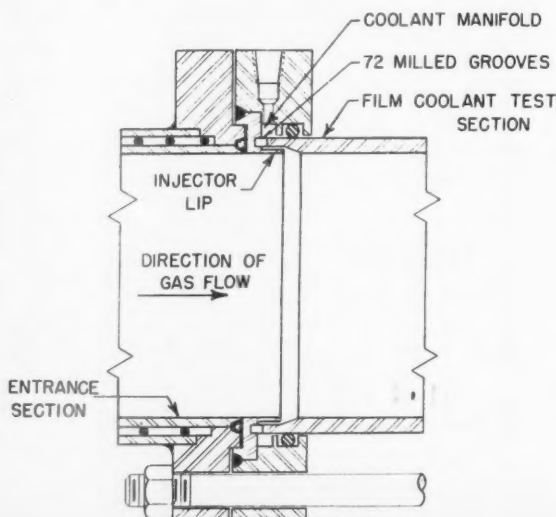


Fig. 8 Film-coolant Injector A

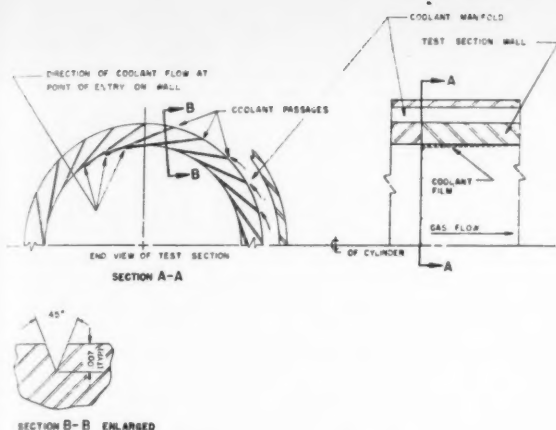


Fig. 9 Film-coolant Injector B

view of one of the grooves is presented as Section B-B. The liquid film coolant, therefore, enters the combustion chamber with a tangential velocity component. It is then swept downstream along the inner surface of the combustion chamber by the hot combustion gases. Experiments with Injector B showed that its inherent profile effectiveness was superior to that for Injector A.

In order to compare the profile effectiveness of Injectors A and B, the heat transfer to the liquid film was calculated by means of [14]. The profile effectiveness was then calculated by means of [6] employing the curve for a smooth surface in Fig. 5 to determine  $\epsilon_s$  and the experimental data presented in Fig. 7. According to those calculations, the value of the profile effectiveness was 50 per cent for Injector A and 63 per cent for Injector B. The relatively low values for  $\epsilon_p$  for both injectors are attributed to unavoidable localized hot spots in the rocket motor due to the characteristics of the main propellant injector.

## 6 Effect of Film Cooling on the Performance of a Rocket Motor

It is desirable to have data regarding the loss in the specific impulse of a rocket motor due to film cooling. For a film-cooled rocket motor developing the thrust  $F$ , the specific impulse  $I$  is defined by

$$I = \frac{F}{\dot{W}_o + \dot{W}_f + \dot{W}} \quad [15]$$

where  $\dot{W}_o$  and  $\dot{W}_f$  are the oxidizer and fuel weight flow rates respectively.

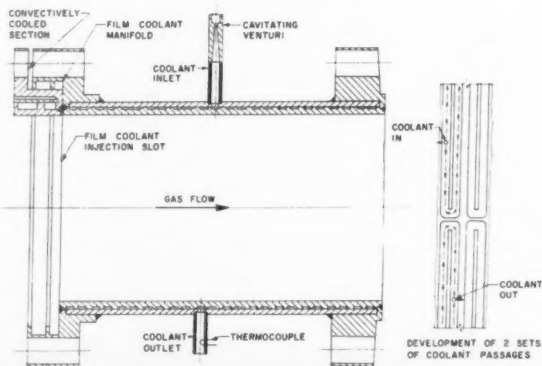


Fig. 10 Convectively cooled, film-cooled test section

The decrease in specific impulse due to film cooling can be determined by operating two geometrically similar rocket motors, one regeneratively cooled and one film cooled, at identical values of combustion pressure and mixture ratio, and measuring their specific impulses. Since the difference in the specific impulses would be small in any case, at most only a few per cent, it is necessary to measure the two values of specific impulses with great accuracy. Since it is difficult to measure them with the required degree of accuracy, few (if any) reliable data on the effect of film cooling on rocket motor performance have been reported. For those reasons a different method was employed in the experiments discussed herein.

Fig. 10 illustrates the film-cooled test section of the rocket motor employed in the experiments. The film-cooled test section is also convectively cooled.<sup>12</sup> It is possible with that

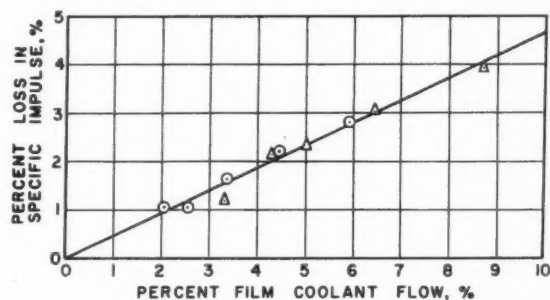
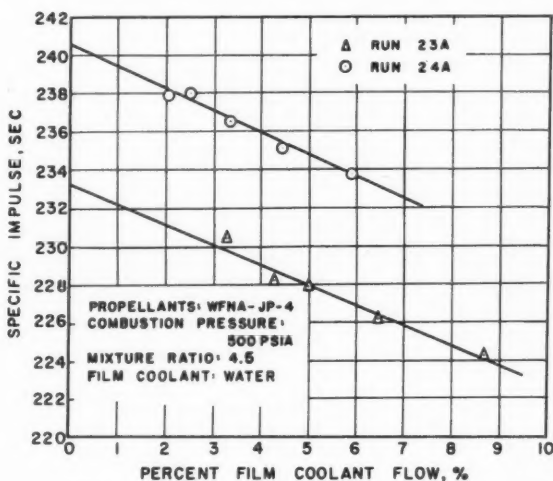


Fig. 11 Effect of film-cooling with water on performance

test section to vary the film-coolant flow rate so that the liquid film will extend different distances down the wall of the test section without overheating occurring in those regimes where there is no film cooling. During an experiment, which lasted approximately 2 min, the film-coolant flow rate was decreased in steps from approximately 0.2 lb/sec to approximately 0.05 lb/sec. Fig. 11(a) presents the specific impulse<sup>13</sup> as a function of the per cent film-coolant flow rate<sup>14</sup>

<sup>12</sup> The test section has twelve convectively cooled sections by means of which the heat transfer to each half-inch section of the test section may be measured. Therefore, the reduction in heat transfer due to film cooling may be measured as a function of length.

<sup>13</sup> Corrected for heat transfer by the method employed in (4).

<sup>14</sup> The per cent film-coolant flow indicates the percentage of the total flow which was introduced into the rocket motor as a film coolant.

for two such runs; each was conducted at a propellant oxidizer/fuel ratio of about 4.5 and a combustion pressure of 500 psia, with water as the film coolant. For each experiment the data are seen to be consistent, although the curves for the two experiments are not in complete agreement with each other; the difference between the two curves represents the experimental error. To obtain an "effective specific impulse" corresponding to zero film-coolant flow, each curve was extrapolated to zero film-coolant flow.

If the per cent error in measuring the specific impulse has a certain magnitude for one value of film-coolant flow, it is reasonable to expect that the same per cent error was present for all values of film-coolant flow during the same experiment. Furthermore, it is reasonable to assume that the effective specific impulse for zero film-coolant flow will also be in error by the same percentage. For any value of film-coolant flow rate, the per cent loss in specific impulse, denoted by  $\delta I$ , is calculated from

$$\delta I = 100 \left( \frac{I_0 - I_{FC}}{I_0} \right) \dots \dots \dots [16]$$

where  $I_0$  is the effective specific impulse for zero film-coolant flow, and  $I_{FC}$  is the specific impulse for any particular value of film-coolant flow rate. It is seen from [16] that when the errors in measuring the specific impulses are consistent, they cancel out when calculating  $\delta I$ .

Fig. 11(b) presents the per cent loss in specific impulse as a function of the per cent film-coolant flow for film cooling with water, as obtained by applying [16] to the two curves of Fig. 11(a). It is seen that the scatter of the data is quite small, indicating that the method employed for obtaining the curve is adequate. Furthermore, only two or three experiments are necessary for determining such a curve.

## 7 Conclusions

The basic considerations pertinent to the film cooling of a circular duct have been presented. It is apparent that before film cooling of rocket motors can be put on a sound engineering basis, additional experimental data are required on  $\epsilon_s$ ,  $\epsilon_p$ , and  $h_f$ . For the present time the required film-coolant flow rate may be estimated by employing Equation [6], where  $\epsilon_s$  is evaluated by means of Fig. 5,  $\epsilon_p$  may be estimated from the values reported in Section 5, and  $h_f$  may be calculated by means of Equation [14] employing the values of  $(f/2)R'$  and  $R$  suggested in Section 4. Additional experiments are needed to determine the effect of film cooling upon the performance of a rocket motor when using reactive film coolants such as jet engine fuel, white fuming nitric acid, etc. Current investigations at the Jet Propulsion Center are concerned with obtaining part of the information.

The semitheoretical method for calculating the required film-coolant flow presented in this paper may be employed for estimating the required film-coolant flow rate for cooling a nozzle (film coolant being injected at one or more locations along the wall of the nozzle). In that case the nozzle is divided into several sections by parallel planes perpendicular to the axis of the nozzle, and each section is analyzed separately.

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# Preliminary Photoelastic Design Data for Stresses In Rocket Grains

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The propellant grain for a solid fuel rocket must be designed with a configuration that maintains the stress concentration factor at the minimum practical value to achieve the maximum resistance to the various mechanical and thermal stresses the grain may undergo during the normal service life of the rocket. In an attempt to establish a systematic approach to this design problem for internal-burning star-perforated grains an investigation was made on the quantitative relationships between the various parameters of grain geometry and the elastic stress distribution in the propellant. Photoelastic stress analyses were made on a family of thick-walled cylinders with symmetrical internal slots of various width, depth and shape covering the range from 2 to 16 slots. Experiments simulating static internal hydraulic loading were made on thin sections of these cylinders. From the data obtained, engineering design charts were prepared showing the maximum stresses developed as a function of the number of slots, the width of the slot, the wall thickness and the shape of the star point. These charts can be used to determine the maximum stress developed in any prescribed configuration within the family investigated providing the properties of the propellant are in the elastic range.

## Nomenclature

- $a$  = radius of the inner surface or locus of star points
- $b$  = radius of the outer surface of the grain
- $d$  = slot width
- $f$  = fringe-value constant of the specimen material
- $K$  = stress-concentration factor or the ratio of the number of fringes  $n$  at any point to the number of fringes which would exist in a thick-walled cylinder of equivalent wall thickness
- $K_i$  = stress concentration at the inner surface
- $K_o$  = stress concentration at the outer surface
- $n$  = number of fringes in the photoelastic patterns
- $N$  = number of star points or slots in the configuration
- $p_o$  = pressure on the outside of the grain (positive inward)
- $p_i$  = pressure on the inside of the grain (positive outward)
- $r$  = radial distance to a point of interest in the cylinder wall
- $t$  = thickness of the photoelastic test specimen
- $W$  = web thickness =  $b - a$
- $\theta$  = angular position coordinate
- $\omega$  = angle between adjacent sides of the star point
- $\rho$  = fillet radius at a star point

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Fig. 1 Typical model of a star-perforated internal burning grain showing the isochromatic patterns caused by simulated internal pressurization

- $\rho_f$  = fillet radius at the corners of a flattened star point
- $\sigma_r$  = radial tensile stress in a star-perforated grain tube
- $\bar{\sigma}_r$  = radial tensile stress in a thick-walled cylinder
- $\sigma_\theta$  = tangential tensile stress in a star-type configuration
- $\bar{\sigma}_\theta$  = tangential tensile stress in a thick-walled cylinder

## Introduction

INTERNAL-burning propellant grains with star-shaped perforations of the type shown in Fig. 1 have been used extensively for solid fuel rockets for at least six years, primarily because they can be used where the grain must be case-bonded or where heat insulation of the motor tube is required to permit the use of lightweight metal parts. Other advantages of this type of grain configuration are broad flexibility in grain design (1),<sup>3</sup> and generally higher loading densities than can be obtained with most external-burning types of propellant charges. However, the internal-burning grain with the complex perforation is subject to the inherent weakness that stress concentrations which occur around the small angles of the star points tend to aggravate any susceptibility of the propellant to mechanical failure. In some situations this can be a critical factor, since handling or firing of the rocket may impose stresses on the grain either as a result of thermal effects (2), particularly where the propellant is bonded to the motor tube, or from momentary pressure differences across the wall or web of the grain.

When handling or firing stresses are imposed on the propellant, the small angles of the star points produce the familiar notch stress concentration effect (3). Under conditions where the propellant is susceptible to failure without plastic deformation, this notch effect can reduce the resistance to a fracture in direct proportion to the value of the stress concentration. For example, in a severe case the maximum stress around a star point (or notch) in a grain with a complex perforation shape may be as much as eight times that which occurs in a simple thick-walled cylinder with an equivalent thickness. Where such a stress concentration is developed, a pressure differential across the web can produce eight times the tensile stress which would occur in a comparable tubular grain. This stress can contribute to mechanical failure of the propellants and consequent malfunction of the rocket motor.

<sup>3</sup> Numbers in parentheses indicate References at end of paper.

To minimize the probability of mechanical failure it is essential to maintain the geometrical stress concentration factors at the lowest practical level, particularly where service conditions are expected that may subject the propellant grain to appreciable unsupported internal pressures. It is also desirable to allow maximum freedom in selecting grain geometry and motor dimensions (4). However, when the design parameters must be balanced to achieve features like maximum loading density and minimum burning time while maintaining an essentially constant burning surface and a reasonable port area through the charge, there is only slight freedom of choice in selecting an ideal configuration to minimize stress concentration.

Previous investigators have used photoelastic techniques to evaluate the stress distributions in certain grain configurations and to show what general types of designs should give the best results (5). The results of this previous work were extremely valuable to the design engineer, but the techniques used made it necessary for him to analyze each shape individually. General guide lines were established for selecting or improving a grain design, but it was not possible to make a quantitative analysis of a series of shapes without direct laboratory observations. In an attempt to provide a more direct approach to grain design, the work described herein was undertaken to establish quantitative relationships between the various parameters of grain geometry and elastic stress distribution in the propellant.

The present paper is devoted to the analysis of stresses resulting from pressure differences across the web or wall of cylindrical internal-burning grains, primarily those with star-shaped perforations (a few exploratory tests were also made on a grain with a square cross section and a square perforation). For most engineering purposes the similarity between the stress patterns produced by internal pressure on unsupported grains and differential thermal expansion in case-bonded grains is sufficient to warrant at least cautious use of the design charts presented for work on the case-bonded units. The case-bonded problem was not studied in the experiments reported<sup>4</sup> but it will be the subject of a separate paper along with an analysis of thermal stresses caused by temperature differences across the web of the propellant grain.

### Application of Photoelastic Techniques

To the best of the authors' knowledge the first published work dealing with a photoelastic study of stresses in propellant grains was that reported by Durelli and Lake<sup>6</sup> which was summarized in a later paper, in which particular emphasis was placed on the experimental techniques (5). Aside from a few minor deviations, Durelli's conclusions and results were employed in the studies covered by this paper.

These studies considered the problem of a long cylindrical hollow tube with a constant-radius external boundary and an arbitrary regularly distributed or periodic internal boundary loaded by uniform internal pressure. Such a problem is known as one in plane strain, which can be treated as an analogous problem of plane stress, which in turn is amenable to photoelastic analysis. It is immediately evident, however, that to impose a uniform internal pressure upon many varying internal contours is a time-consuming experimental problem. One is therefore led to inquire if, for engineering purposes, the same results can be obtained by imposing an external pressure upon the external fixed boundary. Thus the internal configuration could be altered more or less at will while retaining the same experimental setup. This point has been explored by Durelli (5) and answered in the affirmative for a

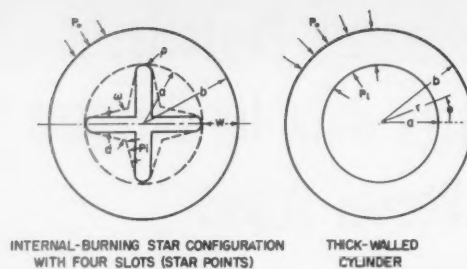


Fig. 2 Typical cross section of an internal-burning star-perforated grain and a cross section of a thick-walled cylinder with equivalent web thicknesses

six-slotted grain. This conclusion was extended to other grain shapes for the work reported herein. This extension is regarded as justified in view of the preliminary nature of the study and of some theoretical analysis of stresses in configurations at the extreme limits considered.

In a thick-walled hollow cylinder subjected to uniform internal and external pressure, as shown by Fig. 2, elementary elastic analysis leads to the following expressions for the radial ( $\sigma_r$ ) and tangential ( $\sigma_\theta$ ) stresses (6)

$$\bar{\sigma}_r = \frac{a^2 b^2 (p_o - p_i)}{r^2 (b^2 - a^2)} + \frac{a^2 p_i - b^2 p_o}{b^2 - a^2} \quad [1]$$

$$\bar{\sigma}_\theta = -\frac{a^2 b^2 (p_o - p_i)}{r^2 (b^2 - a^2)} + \frac{a^2 p_i - b^2 p_o}{b^2 - a^2} \quad [2]$$

where  $p_i$  and  $p_o$  are the positive inside and outside pressures, respectively, and where  $r$  is the radial distance to a point in the cylinder wall. Inasmuch as the axial symmetry of the geometry and loading is such that no  $\bar{\tau}_{r\theta}$  shear stress exists,  $\bar{\sigma}_r$  and  $\bar{\sigma}_\theta$  are principal stresses  $\sigma_1$  and  $\sigma_2$ , and their difference is proportional<sup>6</sup> to the fringe order by the stress optic law (7).

$$\bar{\sigma}_r - \bar{\sigma}_\theta = \frac{2f\bar{n}}{t} = \frac{2(p_o - p_i)}{(b/a)^2 - 1} \cdot \left(\frac{b}{r}\right)^2 \quad [3]$$

Here, then, the fringe order is proportional only to the pressure difference across the wall. This latter fact plus Durelli's conclusion for the six-slotted grain supports a generalized conclusion for engineering purposes in any other configuration which is approximately axially symmetric (independent of the angular position  $\theta$  at any angular cut).

The experimental setup, following Durelli, consisted of a nitrogen-pressurized rubber tube contained on three sides by a steel housing and on the fourth side by the external boundary of the specimen, which was made from (nominal) 1/4-in.-thick sheets of CR-39 transparent resin.<sup>7</sup> After calibrating the gage pressure against effective pressure using thick-walled disk specimens and Equation [3], photographs were taken of the fringe patterns resulting from the various configuration changes. The results contained herein are presented in terms of the ratio of  $n$ , the experimentally determined number of fringes at a point, to  $\bar{n}$ , the number of fringes which would exist in a thick-walled cylinder. Thus we find

$$K = \frac{n}{\bar{n}} = \frac{\sigma_1 - \sigma_2}{\bar{\sigma}_1 - \bar{\sigma}_2} \quad [4]$$

which is a measure of the stress concentration. In normal use, a stress concentration factor is the ratio of a single actual stress to a single nominal stress. However, the previous definition is chosen as more appropriate for this study in accord-

<sup>4</sup> First-order correction factors for the case-bonded grain subjected to a pressure loading or uniform heating are indicated, however.

<sup>5</sup> Published only as an Armour Research Foundation Progress Report.

<sup>6</sup> For the specimens used in these tests,  $2f/t = 308$  psi per fringe.

<sup>7</sup> Columbia Resin Type CR-39, available from Cast Optics Corp., Riverside, Conn.

ance with the remarks upon the sensitivity of the fringes to pressure difference. Actually, at an unloaded boundary one of the principal stresses would be zero, and the two definitions become identical.

### An Unsupported Outer Wall

If, for example, the results are applied to the situation in which only internal pressure is applied and the external boundary is unloaded, the radial stress  $\sigma_r$  would be zero on the outside periphery (indicated by a zero subscript), as would the comparable value  $\bar{\sigma}_r$  in the thick-walled cylinder. Therefore using Equations [3 and 4], the following expression is derived relating the tangential stress to the stress concentration factor

$$\frac{\sigma_{\theta_0}}{p_i} = K_o \frac{\bar{\sigma}_{\theta_0}}{p_i} = K_o \frac{2}{(b/a)^2 - 1} \dots\dots\dots [5]$$

On the internal boundary  $r = a$ ,  $\sigma_r = -p_i$ , and the thick-wall stress is also equal to  $-p_i$ . Hence

$$(-p_i) - \sigma_{\theta_i} = K_i(\bar{\sigma}_r - \bar{\sigma}_{\theta}) = K_i \frac{2(-p_i)}{(b/a)^2 - 1} \left(\frac{b}{a}\right)^2$$

or

$$\frac{\sigma_{\theta_i}}{p_i} = K_i \left[ \frac{2}{1 - (a/b)^2} \right] - 1 \dots\dots\dots [6]$$

Using Equations [5 or 6] the boundary tangential stresses developed in the propellant grain resulting from internally applied pressure can be determined when the stress concentration factors ( $K_i$  or  $K_o$ ) are known. To facilitate this determination the factors

$$\frac{2}{(b/a)^2 - 1} \qquad \frac{2}{1 - (a/b)^2}$$

have been plotted in Fig. 3 as functions of the web fraction  $W/b$ . The experimental determination of the stress concentration factors  $K_o$  and  $K_i$  is discussed subsequently.

### A Case-Bonded Outer Wall

Because the number of fringes depends only upon the pressure difference across the wall, and because the restraint on the outer boundary due to a case or shell of different material introduces no appreciable shear stress  $\tau_{r\theta}$  in the grain but acts essentially as some equivalent (uniform) exterior pressure, it is permissible—to the degree of our present approximation—to assume that the *geometric* stress concentration factors are the same as in the case of an unloaded boundary. Thus if the subscript  $c$  refers to a case-bonded quantity, with the case

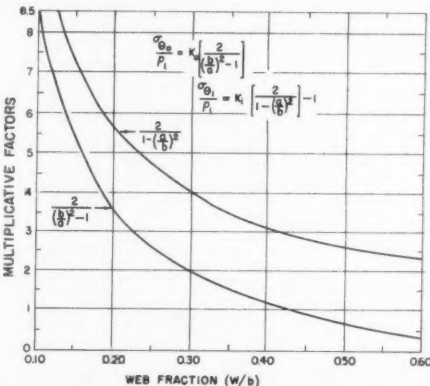


Fig. 3 Multiplicative factors used with the stress concentration when determining actual tangential stresses from applied pressure

thickness,  $h \equiv c - b$ , of material with Young's modulus and Poisson's ratio  $E_c$  and  $\nu_c$ , respectively, we have for the plane stress problem

$$\sigma_{rc} - \sigma_{\theta c} = K(\bar{\sigma}_{rc} - \bar{\sigma}_{\theta c}) \dots\dots\dots [7]$$

For the internally loaded, uniformly thick case-bonded configuration, elementary plane stress analysis leads to

$$\begin{aligned} \bar{\sigma}_{rc} - \bar{\sigma}_{\theta c} = & \frac{2(-p_i)}{\left(\frac{b}{a}\right)^2 - 1} \left(\frac{b}{r}\right)^2 \times \\ & \frac{\left[ 1 + \nu_c + 1 - \nu_c \left(\frac{b}{c}\right)^2 - \frac{E_c}{E} (1 - \nu) \left(\frac{b^2}{c^2} - 1\right) \right]}{\left\{ \left[ 1 + \nu_c + (1 - \nu_c) \left(\frac{b}{c}\right)^2 - \right. \right.} \\ & \left. \left. \frac{E_c}{E} \left[ 1 + \nu + (1 - \nu) \left(\frac{b}{a}\right)^2 \right] \frac{(b/c)^2 - 1}{(b/a)^2 - 1} \right] \right\}} \end{aligned} \dots\dots\dots (8)$$

upon matching the stresses and displacements across  $r = b$  and requiring  $\sigma_{rc}(c) = 0$ . Note that for  $h = 0$  (i.e.,  $c = b$ ) the stress difference becomes identical with [3] for zero external pressure. Inasmuch as for most cases the shell thickness is small compared to the grain radius (viz.  $h \ll b$ ), Equation [8] can be simplified to the approximate relation

$$\bar{\sigma}_{rc} - \bar{\sigma}_{\theta c} \doteq \frac{2(-p_i)(b/r)^2}{(b/a)^2 - 1} \left[ 1 - 2 \frac{(E_c/E)}{(b/a)^2 - 1} \cdot \frac{h}{b} \right] \dots\dots [8a]$$

Because of the displacement interaction between the two materials at  $r = b$ , solutions for the plane strain (cylinder) and plane stress (flat slice) problem are no longer exactly interchangeable. They are, however, simply related when end effects on the cylinder are neglected; using the customary correlation,<sup>8</sup> the  $(E_c/E)$  ratio in [8a] becomes  $(E_c/E) (1 - \nu^2)/(1 - \nu_c^2)$ .

On the internal surface,  $r = a$ , where the pressure is  $\sigma_r = -p_i$ , it is found that the circumferential stress, using [7 and 8a], becomes

$$\frac{\sigma_{\theta ci}}{p_i} = \frac{2}{1 - (a/b)^2} K_i \left[ 1 - \frac{2}{(b/a)^2 - 1} \cdot \frac{E_c}{E} \cdot \frac{1 - \nu^2}{1 - \nu_c^2} \cdot \frac{h}{b} \right] - 1 \dots\dots [9]$$

which reduces to [6], as it should, in the case that there is no shell; i.e.,  $h = 0$ .

It should be emphasized that the  $K_i$  in [9] is the same quantity as given in the design curves; further it may be observed that for the same internal pressure, the internal circumferential stress when the grain is bonded is smaller by the correction factor in the bracket. A similar calculation can be easily carried out for other stress quantities of interest, as well as for the related problem of a uniform heating or cooling of the assembly.

### Parameters of Grain Geometry and Stress Distribution

In investigating the relationships between grain geometry and stress distribution the parameters which were studied included the number of slots (star points), the ratio of slot width  $d$ , or fillet radius  $\rho$ , to the minimum web thickness,<sup>9</sup> and the angle  $\omega$  between adjacent sides of the slot.

<sup>8</sup> See, for example, Timoshenko and Goodier, "Theory of Elasticity," 2nd edit., McGraw-Hill, 1951, p. 408.  
<sup>9</sup> The parameter web fraction  $W/b$  may be used interchangeably with web ratio  $b/a$  inasmuch as  $W/b = 1 - (a/b)$ .

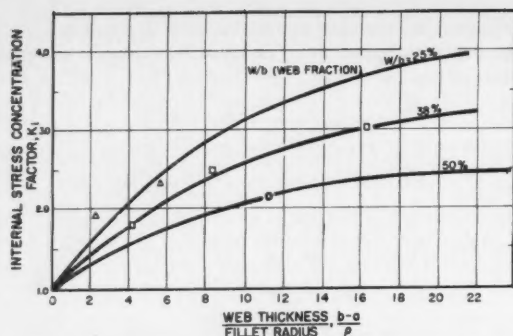


Fig. 4 Experimental data for an eight-slotted grain showing the internal stress concentration factor as a function of web thickness, fillet radius and web fraction

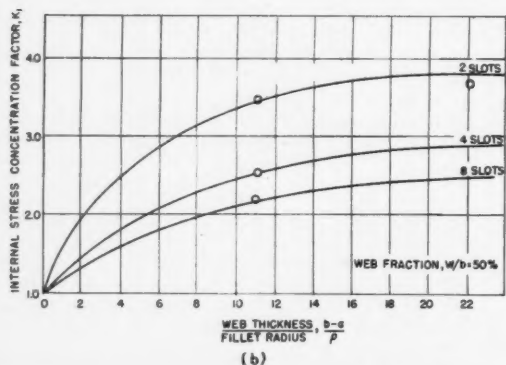
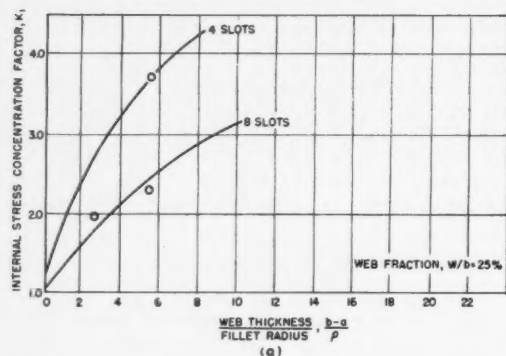


Fig. 5 Experimental data showing the effect of the number of slots on stress concentration factors for two different constant web fractions

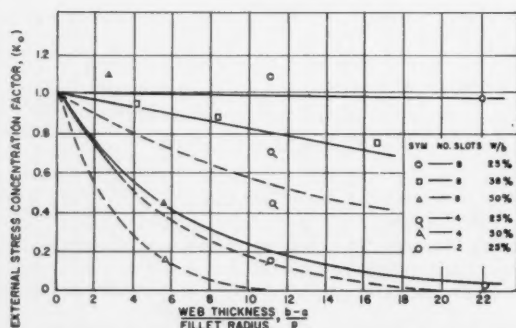


Fig. 6 Experimental data showing the external stress concentration factor as a function of web thickness, fillet radius and the number of slots for various web fractions

Models of the grain designs investigated were machined from sheets of CR-39 resin. These models were subjected to a series of pressures, usually around the external surface, using the experimental apparatus previously described. Stress patterns were photographed using monochromatic polarized light. Fig. 1 shows a typical example of the stress distribution patterns obtained.

In general, the systematic variations in each design parameter were made by successive machining operations on one specimen. For example, the models with the thinnest and shortest star points were prepared first. After the observations on the first form of the model were completed, subsequent designs were prepared by additional machining to remove more material for the longer and wider slots or star points. The actual experimental data are shown in Figs. 4 to 6, expressed as functions of the desired variables in terms of the inside ( $K_i$ ) and the outside ( $K_o$ ) stress concentration factors.

In order to obtain the maximum utility from this very limited amount of data, considerable cross-plotting, interpolation and extrapolation, such as are described by Neuber (3), and by Heywood (8), were used to construct the design charts shown by Figs. 7 to 13. Although the effects of the various design parameters are interrelated in a rather complicated fashion, the following general observations can readily be made by examination of the design charts:

- 1 Stress concentrations vary directly with the web thickness  $W = b - a$  and inversely with the fillet radius  $\rho$ .
- 2 Stress concentrations decrease as the web fraction  $W/b$  increases.
- 3 For a given web thickness and fillet radius the stress concentration decreases as the number of star points is increased.

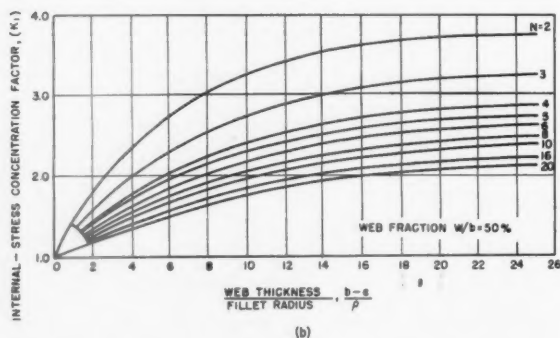
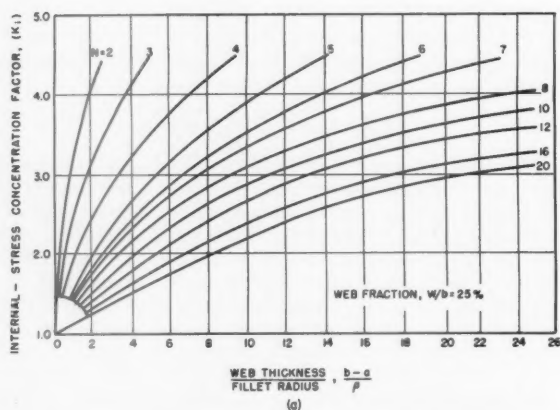


Fig. 7 Design curves showing the effect of web thickness, fillet radius and the number of slots on the internal stress concentration factor for two different constant web fractions

4 Internal stress concentration factors drop rapidly as the star-point angle is increased from 0 to 180 deg.

5 The stress concentration for a configuration with a given web thickness can be decreased by increasing the width of the slot.

From the design curves of Figs. 7 to 13 the geometrical stress concentration factors can be determined for any particular grain shape within the boundaries of the various designs considered. The factor obtained by this method can be used with the multiplicative factors (Fig. 3) to determine the maximum stress in the propellant for any applied internal pressure providing the propellant remains in the elastic region. In addition, for a given configuration the effects of varying any, or a combination, of the major grain-design parameters (i.e., the number of star points, the depth and width of the point or slot and the fillet radius) can be estimated with useful engineering accuracy without experimental observations.

In conjunction with the analysis of the effect of the star-point angle, Fig. 12 is presented to facilitate determination of the stress ratio  $R$  from the values of the angle  $\omega$ , the fillet radius  $\rho$  and the inner radius  $a$ . These curves enable the determination of the stress concentration factor  $K_i(\omega)$  for a particular angle in terms of the factor  $K_i(0)$ , which would apply if the star point was a parallel-sided slot ( $\omega = 0$ ). Conveniently, the data required to determine this latter factor are available in Figs. 7 to 10. The stress concentration factor for the particular angle of interest is then found from the expression

$$K_i(\omega) = 1 + R[K_i(0) - 1] \dots\dots\dots [10]$$

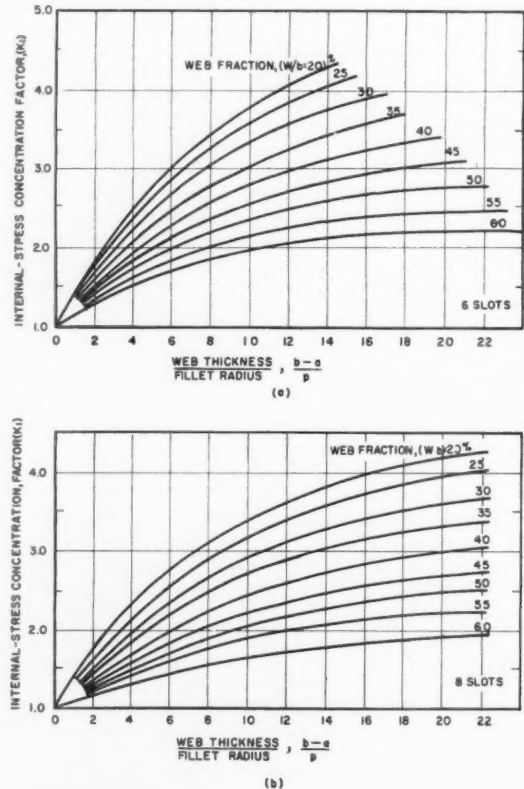


Fig. 8 Design curves showing the effects of web thickness and fillet radius on the internal stress concentration factor as a function of web fraction for six- and eight-slotted configurations

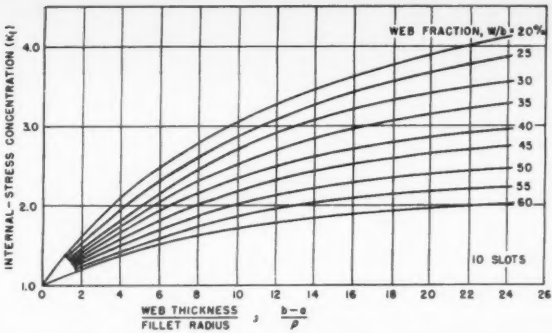


Fig. 9 Design curves showing the effects of web thickness and fillet radius on the internal stress concentration factor as a function of web fraction for ten slotted configurations

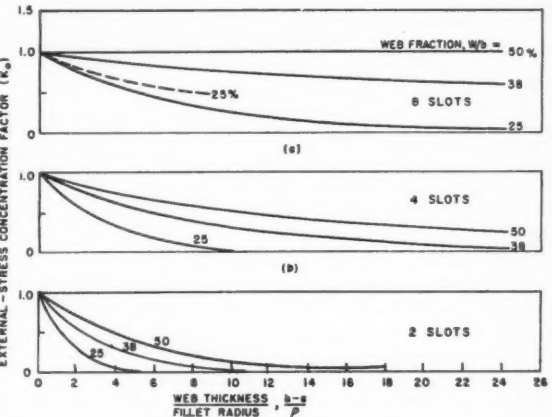


Fig. 10 Design curves showing the effects of web thickness, fillet radius and web fraction on the external stress concentration factors for various numbers of slots

Fig. 12 was prepared using a modified parameter for Heywood's (8) suggested curves for predicting the effect of opening angle  $\omega$  from a known value at  $\omega = 0$ . Since these curves were constructed for shallow notches, they are not necessarily applicable to the deep slots or star points typical of the grain designs considered for the present work. On the other hand, the experimental data, although not definitive, suggest that the same curve could be used as a first approximation by replacing Heywood's parameter  $\rho/h$ , where  $h$  is the depth of the shallow notch, by  $\rho/a$ . This substitution was used to construct the design curves given in Fig. 12. This matter should perhaps be the subject of further study.

Another variation which is often considered in grain design is the widening of the slot while flattening the point. In a preliminary study of this variation, one rather limited test was made to evaluate the effect of the interrelated variables of slot width and fillet radius on the circumferential stress midway between the sides of the slot. Fig. 13 shows (a) the test data obtained for an eight-slotted specimen with 25 per cent web fraction, and (b) the suggested design chart. As might be suspected from St. Venant's principle, the maximum variation occurs within the range  $1/2 \leq \omega/d \leq 2$ .

One other shape briefly examined was a square internal contour with either a square or circular external contour. For some of the tests made on this grain shape, the pressure was applied internally as well as externally to check the consistency of the data. For the internal pressure tests a rubber tube was again used as the loading device with a central filler contour also made of CR-39. Two thick parallel Lucite

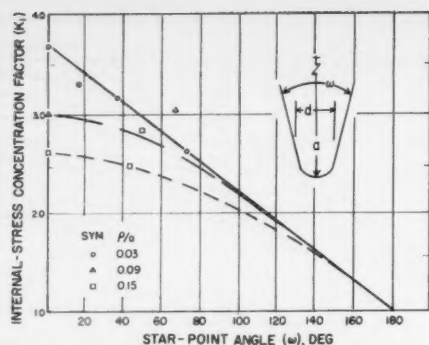


Fig. 11 Experimental data showing the effect of the star point angle on the internal stress concentration factor

plates—to all intents and purposes insensitive to polarized light—were installed with suitable spacers on opposite sides of the specimen to absorb transverse pressure. The assembly was completed by bolts through the central portion and aligned along the axis of the specimen.

Fig. 14 shows the test data and design curves obtained for this configuration. The tangential stress in the fillet at the inside boundary is shown as a function of the web fraction. The lower curve was estimated from Figs. 9 and 10 assuming a four-slotted grain with  $\omega = 90$  deg. It can be seen that the shape of circular contour has significantly lower stresses. Experiments also showed that when the four outside corners of the square configuration were rigidly supported the stress patterns were essentially unaltered over the range tested.

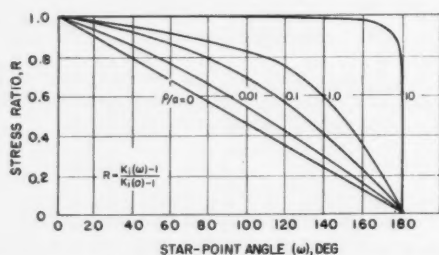


Fig. 12 Tentative design curves for determining the effects of the star point angle on the stress concentration

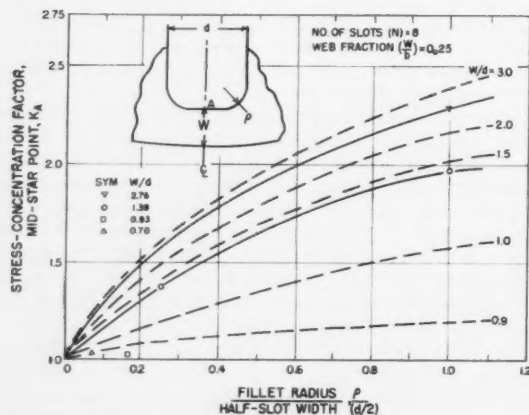


Fig. 13 Design curves for determining the effects of slot width and fillet radius on the stress concentration at the mid-point of a slot

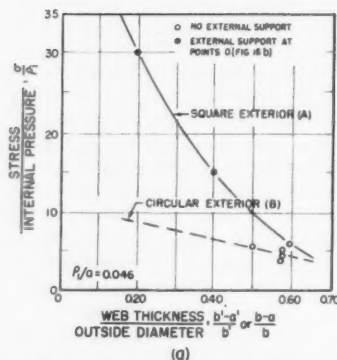
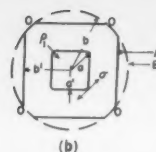


Fig. 14 Some effects of the principal design parameters on stress concentration in a square grain

## Conclusions

In summary, it can be said that the results of this investigation substantiate what the designer might reasonably predict since they indicate that stress concentrations in rocket grains can be reduced by: (a) Reducing the web thickness, (b) increasing the fillet radius, (c) increasing the web fraction, (d) increasing the number of star points, (e) increasing the star-point angle and (f) increasing the width of the slot (or star point).

Further, by charts, stress concentrations and actual stress values can be rapidly determined as functions of the major design parameters and the applied pressure without the need for direct measurement. As a result, within the limits imposed by other considerations, the grain designer should be able to select a balanced configuration to minimize stress concentrations for the first prototype of a new development.

Finally, it should be emphasized that the purpose of the investigation was to obtain some data which would provide a basis for quantitative design. It should be obvious that accuracy suffers whenever a great deal of cross-plotting and extrapolation is employed to extend a limited amount of test data. Nevertheless, it is believed that for the variations examined, in the absence of specific information to the contrary, the suggested design charts can be used until related loading and environmental conditions become better defined.

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# Field Transportation of Concentrated Hydrogen Peroxide

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In the twelve years since concentrated (90 per cent) hydrogen peroxide was first manufactured, a great deal of effort has been expanded toward the development of methods of handling and transporting this propellant with maximum simplicity and safety. This paper discusses the methods of handling hydrogen peroxide commercially, outlines the requirements of hydrogen peroxide tactical service vehicles, and presents examples of field service units which embody the principles outlined.

## Introduction

NINETY per cent hydrogen peroxide is a liquid propellant which has found many military uses, both as a monopropellant and as an oxidant in bipropellant systems. As a monopropellant it has been used chiefly for gas generation to power turbo-pumps for pumping other propellants. Typical of this is the use of  $H_2O_2$  in the Viking rocket. As an oxidant it provides 0.423 lb  $O_2$ /lb of 90 per cent  $H_2O_2$ . Examples of the use of  $H_2O_2$  as an oxidant in two fluid systems are the German torpedo and Walter Cycle submarine developments. The propellant characteristics of  $H_2O_2$  have been discussed in papers by Davis, Bloom, and Levine (1)<sup>3</sup> and by Bellinger and others (2). A discussion of certain items of equipment developed or tested for handling  $H_2O_2$  was presented by Davis and Keefe (3).

Present methods of commercial shipment of high strength hydrogen peroxide are satisfactory but for tactical purposes these methods do not have the flexibility required for field support or military operations. Therefore several tactical vehicles for the transportation of high strength hydrogen peroxide were designed and constructed with the use of readily available commercial equipment.

In this paper we wish to discuss the methods for the field transportation of concentrated hydrogen peroxide with emphasis on the recent development of these  $H_2O_2$  transfer trucks.

## Methods for the Commercial Transfer and Handling of $H_2O_2$

The present methods for the commercial shipment of concentrated  $H_2O_2$  are drum shipments by rail or truck, railway tank car and special  $H_2O_2$  tank trucks as shown in Fig. 1.

### Drums

Aluminum alloy 1060 drums, type 42D, are approved by the Interstate Commerce Commission for shipping  $H_2O_2$ . Single-headed drums are used for full carload shipments and the double-headed drums for less than carload shipments. All drums are provided with top filling and venting connections. The vent opening is covered with a slit plastic disk that opens at about 3-psi differential pressure for venting, but prevents splashing and spillage of  $H_2O_2$  if the drum is accidentally placed on its side.

Presented at the ARS Fall Meeting, Buffalo, N. Y., Sept. 24-26, 1956.

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<sup>3</sup> Numbers in parentheses indicate References at end of paper.

Unloading the drum is usually accomplished by a siphon. Pressurizing the drum is *never* recommended.

The drums are normally filled with 250 to 300 lb of 90 per cent  $H_2O_2$  which leaves approximately one quarter of the volume empty to reduce the possibility of spillage. Ninety per cent  $H_2O_2$  can be shipped and stored for long periods of time with no appreciable loss in concentration. A check test on ten drums which had been in transport and storage for periods up to three years showed an average loss in concentration of 0.9 per cent per year.

### Railway Cars

Railway tank cars for  $H_2O_2$  service are available in 4000, 6000, and 8000 gal capacities. These cars are made of 1060, 5652, and 5254 aluminum alloys; 5652 and 5254 alloys have higher tensile strength than 1060 aluminum and their use affords a saving in weight.

Each car is equipped with a dome on which an inspection manhole is mounted. The outlet connection and a vent are also located on the dome. The cars are unloaded and filled through the top connection.

All  $H_2O_2$  storage tanks are provided with filter vents to prevent pressure building up in the tanks in the event of decomposition of the hydrogen peroxide. Vents are protected by means of a filter stone to prevent dirt and dust from the atmosphere contaminating the hydrogen peroxide and causing decomposition. In the event of gross contamination, dilution and dumping of the contents is mandatory since the decomposition reaction is exothermic and the vents will not be able to prevent pressure buildup in the containers.

### Tank Truck Trailer

The 2500-gal  $H_2O_2$  tank truck trailer is normally used for the commercial transfer of  $H_2O_2$  concentrations up to 50 per cent, but is built of materials suitable for the handling of 90 per cent  $H_2O_2$  as well. The trailer is designed for use with standard tractors. It is equipped with a gasoline-powered aluminum self-priming pump and carries several lengths of suction and discharge hose. This makes the trailer a self-contained unit that can be loaded or unloaded without the tractor.



Fig. 1 Commercial methods for shipment of hydrogen peroxide

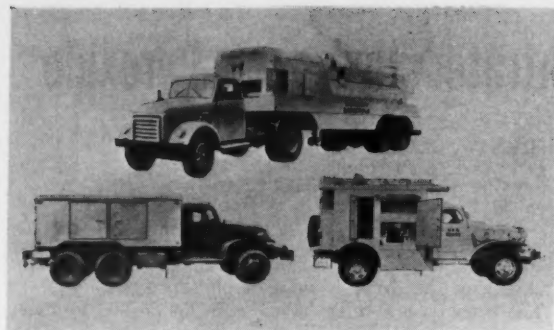


Fig. 2 Field transfer trucks for shipment of hydrogen peroxide

### Requirements for Tactical Hydrogen Peroxide Service Units

The requirements for a tactical hydrogen peroxide service unit are of two types: those which are general to any tactical service vehicle and those which are specific to a hydrogen peroxide unit. The general requirements for a tactical vehicle include the following:

- 1 The vehicle should be ruggedly constructed for possible on-and-off road use. Only a minimum amount of simple maintenance should be required to provide satisfactory performance.
- 2 Provisions should be made for operation under adverse climatic and visual conditions. Lights should be provided for night operation. Insulation and a heating system should be provided to prevent freezing of the truck water supply and critical system components.
- 3 Operation of the unit should be simple and require no more than a crew of two semiskilled men.
- 4 In so far as is practical, spare parts and supplies for the entire unit should be available in military stores.
- 5 The unit should be self-contained. Power for opera-

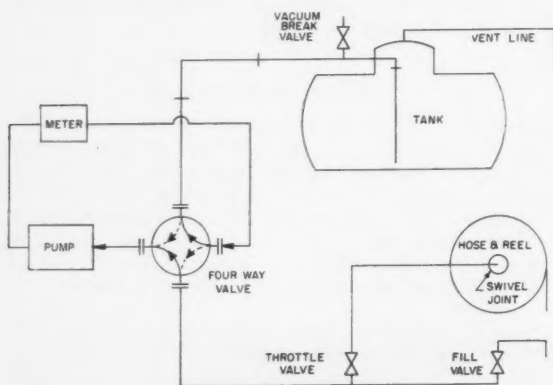


Fig. 3 Hydrogen peroxide flow system

tion of the transfer pumps and for generation of electricity for night lighting and operation of the electrical components should be provided by the main truck engine. Provision should be made for attendant functions such as a water supply as discussed below.

Requirements which are specific to a hydrogen peroxide unit include the following:

- 1 All parts which will contact the hydrogen peroxide should be made of suitable materials. In general, the storage tank and piping should be made of high-purity aluminum or selected aluminum alloys. System components such as the pump, meter and valves should be made of selected aluminum

alloys. A listing of suitable materials is given in a recent publication of the Bureau of Aeronautics (4).

2 The design of each component in the system should be carefully selected. Designs incorporating areas, such as internal threads, where the hydrogen peroxide might be trapped must be avoided. Hydrogen peroxide decomposes at a continuous slow rate evolving oxygen gas which must be vented. Simple and smooth surfaces are desirable. The decomposition of hydrogen peroxide is substantially a heterogeneous reaction and generally varies directly as the ratio of the wetted surface to liquid volume. Sufficient information is available today on a wide variety of acceptable components so that pumps, valves, meters and other components of proved design can be selected.

3 The hydrogen peroxide system should be installed to provide simple and foolproof operation. The use of three- and four-way valves will often eliminate the possibility of  $H_2O_2$  being accidentally trapped between two shut-off valves. Any place where  $H_2O_2$  might become trapped, such as might happen due to an error in the proper setting of valves, should be provided with a safety vent. The system should be designed so it may be completely drained prior to disassembly or replacement of a component.

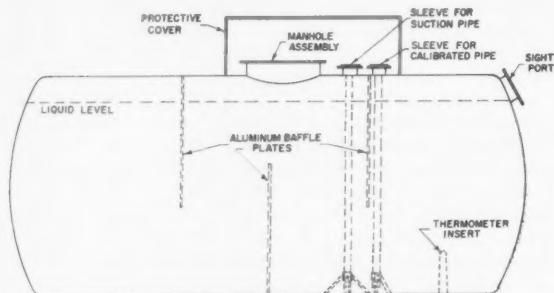


Fig. 4 Hydrogen peroxide storage tank

4 The system should be designed to prevent contamination of the hydrogen peroxide. The storage tank vent should be equipped with a filter to permit breathing and venting of the evolved oxygen gas and yet prevent introduction of dirt from the air. The proper selection of valve designs and pump sealing glands will prevent entry of dirt along valve stems or pump shafts. The complete system should be cleaned and passivated for  $H_2O_2$  service. Once passivated, it will remain passive as long as the system is used with  $H_2O_2$  and should not require periodic passivation. Once removed, hydrogen peroxide should not be returned to the storage tank. If the  $H_2O_2$  has been transferred to a use tank, such as an aircraft tank, and this tank must be defueled, the  $H_2O_2$  should be discarded.

5 Provision should be made for a water supply system for safety showers and for washing down any  $H_2O_2$  spillage. This water tank should generally be of 50 to 100 gal minimum capacity. Provision should also be made for tools and safety equipment, including gloves, eye goggles, aprons, boots and disabled vehicle flares.

6 A device to indicate satisfactory stability of the  $H_2O_2$  is desirable. Temperature alarms are presently used to measure the  $H_2O_2$  temperature and give a visual and sound alarm in the event of excessive temperature rise due to self-heating of the  $H_2O_2$  resulting from contamination. Work is presently being done to develop a device to indicate gas evolution and activate an alarm in the event of  $H_2O_2$  contamination.

### Examples of Tactical Service Units

Examples of tactical service vehicles are shown in Fig. 2. A typical schematic drawing for the  $H_2O_2$  flow system for such

(Continued on page 677)

# Technical Notes

## Time Dilation in Space Flight

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IN THE December 1956 issue of *JET PROPULSION*, Kurt R. Stehling presented in "Space Flight Notes" a review of opinions and arguments concerning time dilation.

Whether one believes that time dilation is true or not, the following facts should be realized.

The theory of relativity may be divided into the special theory and the general theory. The special theory of relativity is restricted to intercomparison of measurements made by observers which have a uniform velocity with respect to each other in a gravitation-free part of space. The time dilation is a direct consequence of the two postulates of the special theory of relativity. However, the formula

$$\Delta t' = \Delta t \sqrt{(1 - v^2/c^2)} \dots \dots \dots [1]$$

where  $\Delta t$  and  $\Delta t'$  are time intervals on earth and in the space ship, respectively, applies only to time intervals during which the space ship has the constant velocity  $v$  or  $-v$  with respect to earth. If the time for acceleration and deceleration of the space ship is very short as compared with the total travel time, Equation [1] gives an approximation for the relation between the elapsed times  $\Delta t$  and  $\Delta t'$ .

The clock paradox would arise if the space ship were considered as being at rest and the earth as moving; formal application of the special theory then results in the relation

$$\Delta t = \Delta t' \sqrt{(1 - v^2/c^2)} \dots \dots \dots [2]$$

The mistake made is that with the rocket motor working, the space ship can only be considered at rest if a gravitational force field is applied that cancels the rocket thrust and that accelerates the earth. But then the special theory cannot be applied, so that the conclusion, relation [2], is false. Therefore, there is no paradox.

The general theory of relativity is based on the relativity of all kinds of motion and includes gravitational effects. Clearly then, to describe the time dilation in terms of a coordinate system in which the space ship appears at rest and the earth is moving, the general theory of relativity is needed. According to the general theory, the rates of two identical clocks in a gravitational field differ by an amount  $\Delta\phi/c^2$  where  $\Delta\phi$  is the difference in gravitational potential for the two clocks and  $c$  is the velocity of light.

Tolman shows<sup>2</sup> that if the space ship is considered at rest and with the gravitational effect taken into account, the time dilation agrees with Equation [1], at least for small  $v$ . There is no reason to believe that a contradiction would arise for more general cases.

Time dilation is a straightforward result of the hypothetical basis of the theory of relativity. Therefore, we can only discard it by rejecting one or more of the hypotheses of relativity. All experiments concerning macroscopic phenomena which would distinguish between Newton mechanics and relativistic mechanics so far have yielded results that confirm

the relativistic theory. Therefore, until experiments prove otherwise, the theory of relativity is a satisfactorily precise description of macroscopic mechanics. But then, there is no reason to doubt the existence of time dilation.

## On the Slowing Down of Time

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THE old and fascinating controversy of the slowing down time for a moving clock, which nowadays is stirring great interest because of its possible—though rather abstract—astronautical consequences,<sup>1</sup> has been recently reported in *JET PROPULSION*.<sup>2</sup>

We would like to point out here that in 1918, in reply to a similar debate, Albert Einstein himself, the creator of the theory of relativity, gave an easy and clear-cut answer to the problem and to the objections put to him.<sup>3</sup> His answer to the question whether a moving clock is left behind a fixed clock is, "Yes." And he regrets that a number of physicists do not share this inescapable conclusion: "Mit Bedauern habe ich gesehen, dass einige Autoren, die sonst auf dem Boden der Relativitätstheorie stehen, diesem unvermeidlichen Ergebnis ausweichen wollten."

His line of reasoning, expounded in a pleasant form of a dialogue between a critic and a relativist physicist, goes, briefly, as follows. A clock A (say) stays at rest in an inertial frame of reference, i.e., with respect to the fixed stars. Another clock, B, goes from A to another point far away, then turns and goes back. The theory of special relativity tells us with certainty that B, on arrival back at the starting point, records a shorter time than the other clock A. One, however, may object that the same considerations should apply for an observer moving together with the clock B: He should conclude that the clock A—which now from his point of view does move—is slower instead, which is contradictory. This argument is wrong. One cannot apply to the new frame the same consideration as before, because it is not an inertial frame, i.e., does not move uniformly with respect to the fixed stars. To such frames special relativity does not apply, and one has to call for the theory of general relativity. By the latter one can prove that any gravitational field has the same effect on clocks as the uniform motion, that is, slows them down. Now during the two intervals in which the clock A moves uniformly, A is left behind the other clock B; but while the clock A is turning, the observer experiences a field of force, since he is really changing direction of motion. This field of force, perfectly equivalent to a gravitational field, causes the clock B to slow down with respect to the other one, just the right amount required to agree with the previous result.

<sup>1</sup> We may quote here only a recent debate in *Nature*, vol. 167, 1951, p. 680; vol. 177, 1956, p. 782; and a paper by Sänger, E., "Photon Rockets," *Aero Digest*, vol. 73, 1956, p. 68.

<sup>2</sup> Stehling, K. R., "Space Travel and Relativity," *JET PROPULSION*, vol. 26, Dec. 1956, p. 1105.

<sup>3</sup> Einstein, A., "Dialog über Einwände gegen die Relativitätstheorie," *Die Naturwissenschaften*, vol. 6, 1918, p. 697.

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<sup>1</sup> Senior Engineer, Supersonics Group.

<sup>2</sup> Tolman, R. C., "Relativity, Thermodynamics and Cosmology," Oxford, 1934.

The experimental verification of this so-called "paradox" is not as far off as it would seem. The time dilatation factor

$$\sqrt{1 - v^2/c^2}$$

for the velocity attained by the artificial satellite is of the order of  $1 - 3.5 \cdot 10^{-10}$ ; now the precision of one part in  $10^{10}$  is quite within the reach of modern atomic clocks.<sup>4</sup>

<sup>4</sup> For a detailed discussion, see Singer, S. F., *Physical Review*, vol. 104, 1956, p. 11.

## On the Hazards of Concentrated Hydrogen Peroxide

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IN THE March 1957 issue of *JET PROPULSION* (pp. 296-298) there is an article on Patrick Air Force Base in which several references were made to hazards involved in handling concentrated hydrogen peroxide. As presented, these hazards seem to be much greater than those of other monopropellants and oxidants, although experience has shown that they are not. We would like to point out wherein the information is in error or has exaggerated the situation.

In the photograph on page 297 a technician is shown checking a drum of Becco 90 per cent by weight hydrogen peroxide to determine whether or not this drum was heating up from  $H_2O_2$  decomposition. The photograph is misleading in several respects. First, it is unnecessary for the man to wear face shield, apron and rubber gloves merely to check the drum temperature; only eye protection with close-fitting goggles is deemed necessary. During unloading of a drum this protective clothing is desirable in case of accidental spillage. If this protective clothing were always necessary, we could not ship drums of concentrated  $H_2O_2$  freely on public carrier under white label regulations. Secondly, the man could not adequately detect temperature rise while wearing a rubber glove on his hand. Thirdly, hydrogen peroxide does not "ignite"; contrary to the caption it actually is a non-flammable material.

At times pressure rupture of drums has occurred. However, in every case brought to our attention it has been proved that  $H_2O_2$  was removed and returned to the drum and hence probably contaminated, a practice which is contrary to Becco's written instructions. Of the many thousands of drums of concentrated hydrogen peroxide that Becco has shipped since first manufacturing this propellant in 1944, no unopened drum with the Becco seal intact has ever suffered contamination sufficient to attain self-heating of the  $H_2O_2$  with resultant drum rupture.

The article indicates (p. 298) that hydrogen peroxide must be treated "very tenderly." As mentioned above this propellant has been shipped on common carrier for years under corrosive liquid regulations only. This cannot be classified as "tender" handling. Admittedly, careless handling of hydrogen peroxide, especially when the drums are opened for removal of liquid, might result in introducing contamination into the drum, with ensuing trouble. However, a cognizance of the instructions for handling, and reasonable attention to the need for cleanliness, has been shown to be adequate for safety as evidenced by the fact that millions of pounds of 90 per cent  $H_2O_2$  have been produced, shipped and consumed in the United States since 1944.

It is stated that a "large speck of dust" will cause a dangerous chain reaction in  $H_2O_2$  leading to a violent explosion. One could hardly envision commercial production and handling

of a material that was sensitive to the effect of a "large speck of dust." Self-heating of  $H_2O_2$  containers was discussed in a paper by E. S. Shanley of Becco, published in *Industrial and Engineering Chemistry*, vol. 45, July 1953, p. 1520. Referring to Table I and Fig. 1 of this paper, it will be noted that at an ambient temperature of 25 C (77 F) 90 per cent  $H_2O_2$  in a standard 30-gal shipping drum with a surface to volume ratio of 2.5 sq ft/cu ft, would have to decompose at a rate in excess of 50 per cent per year to attain self-heating. We know of no material of sufficient catalytic activity to cause a decomposition rate of 50 per cent per year when present in minute quantities in 25 gal of 90 per cent  $H_2O_2$ . In fact, to intentionally contaminate  $H_2O_2$  for drum safety tests, we have found it necessary to use gram quantities of a specially prepared highly active catalyst to obtain a high rate of  $H_2O_2$  decomposition. History shows 90 per cent  $H_2O_2$  in drums decomposes normally at a rate of approximately 1 per cent per year under uncontrolled storage conditions.



Fig. 1 Becco standard 30-gal aluminum  $H_2O_2$  shipping drum after 15 grams of dynamite were set off in 20 gal of 99.5 per cent  $H_2O_2$  at 160 F

We think that the above discussion should indicate that handling hydrogen peroxide is not "touch and go." It is not detonable, as shown in tests reported by Greenspan and Shanley in *Industrial and Engineering Chemistry*, vol. 39, 1947, p. 1536. Further, no untoward results occurred in more recent tests at Becco in which 15 gm of dynamite were set off in 20 gal of 99.5 per cent  $H_2O_2$  heated to 160 F in a 30-gal standard, pure aluminum shipping drum. Fig. 1 shows a photograph of the drum after this test.

One does not consider gasoline a "touch and go" material; however, gasoline vapors normally present over the liquid can be readily ignited by a spark or a match without any warning. Hydrogen peroxide is very much less sensitive than this and always gives a warning of self-heating.

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An Effect of Carbon in an Adiabatically Expanded Gas Stream

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WHEN one is dealing with monopropellants which contain large amounts of carbon, the possibility of observing decreased performance as a direct result of carbon in the gas stream must be considered. Carbon recovered from such propellants has an extremely small bulk density, particle size and, of course, a relatively large internal volume. The possibility of the large internal volume communicating with the external surface through numerous, small, pore-like openings must also be considered.

When one mole of monopropellant decomposes to 3 moles of hydrogen and 5 moles of carbon of bulk density 0.03 (H<sub>2</sub>O = 1.0) at 50 atm and 2460 R, approximately 30 per cent of the gas is found in the internal voids of the carbon. Hydrogen gas is not adsorbed in these voids and the gas pressure within the voids is the chamber pressure. During flow through the divergent section of a nozzle, gas must escape from the internal voids in order to be available to do work. The small, pore-like openings would, of course, slow this process considerably, the net result being that all hydrogen gas is not available for work. The expansion process through the pores, per se, of course does no useful work. Since the pore openings are oriented at random, escape of gas during expansion could conceivably create small areas of shock at the pore openings which would result in a further loss of performance. In order to calculate  $\gamma$  for the solid-gas mixture without gas being trapped within the pores during expansion, we define the  $\bar{C}_p$  of the solid-gas mixture as

Cp\_bar = n\_g C\_p^g + n\_s C\_p^s [1]

where  
Cp\_bar = mean specific heat at constant pressure of mixture  
n\_g = moles of gas  
n\_s = moles of solid  
C\_p^s = specific heat at constant pressure of solid  
C\_p^g = specific heat at constant pressure of gas

The solid is assumed to be in thermal equilibrium with the gas and not to lag.

The mean specific heat  $\bar{C}_v$  of the mixture at constant volume is

Cv\_bar = n\_g C\_v^g + n\_s C\_v^s [2]

where  
C\_v^g = specific heat of gas at constant volume  
C\_v^s = specific heat of solid at constant volume

But  
C\_v^g = C\_p^g - R [3]

where  
R = molar gas constant  
and  
C\_p^s = C\_v^s [4]

Using [1-4],  $\gamma$ , the specific heat ratio of the solid-gas mixture, is

gamma = Cv\_bar / Cp\_bar [5]

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For the same system where some gas is trapped within the voids

Cp\_bar = n\_g^1 C\_p^g + n\_s C\_p^s + n\_g^2 C\_p^g [6]

where  
n\_g^1 = moles of gas external to voids  
n\_g^2 = moles gas within voids  
C\_p^g = specific heat of gas within voids at constant pressure  
and

Cv\_bar = n\_g^1 C\_v^g + n\_s C\_v^s + n\_g^2 C\_v^g [7]

where  
C\_v^g = specific heat of gas within voids at constant volume

When the mixture is expanded through the nozzle we assume that gas within the voids does not escape and

C\_p^g = C\_v^g [8]

Then

gamma = Cv\_bar / Cp\_bar [9]

When  $n_g^1 = n_g$ , i.e., the gas pressure within the voids is always equal to the bulk gas pressure during expansion, Equation [5] is the same as [9].

Table 1 Calculated temperature fall and thrust available from an adiabatically expanded gas stream containing gases trapped in carbon

Moles H <sub>2</sub> available	ΔT (°R)	% Thrust
3.0	910	100
2.7	845	96
2.5	798	93
2.3	753	90
2.0	674	84

For the system 5C + 3H<sub>2</sub> at 2460 R, expanded through a pressure ratio of 40, the temperature fall will depend on the amount of gas available for expansion; viz, external to the voids of the solid phase. Table 1 has been constructed for varying amounts of gas available; unavailable gas is assumed to be trapped within the voids and does not escape during expansion. The exit temperature was calculated from the expression.

T\_e / 2460 = (40)<sup>(gamma-1)/gamma</sup> [10]

The specific heat ratio  $\gamma$  was calculated from [9] at the chamber temperature (2460 R) and was assumed to be independent of the temperature fall. The last column indicates the square root of the fractional enthalpy change or amount of thermal energy that is converted to thrust in an adiabatic expansion as compared with optimum, and was calculated from the expression

ΔH = Cp\_bar ΔT [11]

where  
Cp\_bar = n\_g^1 C\_p^g + n\_s C\_p^s + n\_g^2 C\_p^g

Verification of the postulates put forward here would seem to come from an experimental examination of chamber and exhaust temperatures.

# An Approximate Specific Impulse Equation for Condensable Gas Mixtures

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SINCE the specific impulse of rockets is calculated directly from the enthalpies and velocities in the nozzle, such questions as the attainment of thermal, phase and velocity equilibrium by condensed particles are of great importance. The purpose of this note, however, is to study a relatively simple equilibrium model of the situation and reserve such kinetic problems as the above for future consideration. Specifically, it will be assumed that the equilibrium vapor-liquid and liquid-solid phase relations are followed, that the condensed particles essentially reach the temperature and velocity of the gases, and that radiation from the particles is negligible. This model has received several treatments in the past (1, 2, 3)<sup>2</sup> but attention has usually been confined to flow of solid (or liquid) particles *without* phase changes. Altman and Carter (2) have briefly considered flow with condensation but confined their attention to expansion of pure vapors and the criteria for the occurrence of condensation or evaporation on expansion. Physically realizable conditions will always result in condensation for systems of interest in rocketry. The previous work will be discussed after formulas are derived for the general case of flow with condensation and solidification.

The ideal specific impulse  $I_{sp}$  may be given fairly generally in terms of the difference in enthalpy of chamber and exhaust

$$I_{sp} = \frac{v_e}{g} = \frac{1}{g} \sqrt{2(h_c - h_e)} = \frac{1}{g} \sqrt{2 \frac{(H_c - H_e)}{\bar{M}}} \dots [1]$$

where  $v_e$  is ideal exhaust velocity;  $g$ , acceleration of gravity;  $h$ , enthalpy per unit mass;  $H$ , molar enthalpy;  $c$ , chamber conditions;  $e$ , exhaust conditions; and  $\bar{M}$ , average molecular weight over all species and phases. To an approximation which is usually well within the accuracy of the available thermal data, one can choose an average molar heat capacity  $\bar{C}_p$  and average heats of fusion  $\Delta H_f$  and vaporization  $\Delta H_v$ . Equations [1] may then be written

$$I_{sp} = \frac{1}{g} \sqrt{\frac{2}{\bar{M}} [\bar{C}_p(T_c - T_e) + (y_{gc} - y_{ge})\Delta H_v + y\Delta H_f]} \dots [2]$$

where  $y$  is the mole fraction of condensable species;  $g$ , gas phase;  $l$ , liquid; and  $s$ , solid. Thus

$$y_e = y_{gc} + y_{lc} = y \dots [3]$$

The phase change heats in Equation [2] are frequently omitted in impulse calculations with condensation. The above equations and those following are written for only one (nonreacting) species, but the treatment could be (formally) extended to other condensed species. For nearly all cases of interest  $y_{gc} = 0$  and  $y_{gc} \gg y_{ge}$ . If  $T_e$  is greater than the melting point of the condensable species, the fusion term is, of course, absent.  $\bar{C}_p$  is given by

$$\bar{C}_p = \sum_{i,s} y_i \bar{C}_{pi} \dots [4]$$

where  $\bar{C}_{pi}$  is a suitable average with respect to temperature.

With the chamber conditions given, the problem, as usual, is to find  $T_e$ . This could be done rigorously by a trial and error process involving the equality of the entropy at chamber and exhaust, and suitable assumptions regarding the exhaust composition. However, an impulse equation in closed form

can be derived for flow with condensation (for frozen chemical reaction) as follows: Consider the nozzle expansion as taking place through an isothermal expansion to  $P_e$ , followed by a constant pressure cooling at  $P_e$ , the exhaust pressure. The entropies of the two steps are opposite in sign and may be equated, since the over-all entropy change is zero.

1. Isothermal expansion to  $P_e$ .

$$\Delta S_g = -\sum \frac{x_{ig}}{M_{ig}} R \ln \frac{P_e}{P_c}, \Delta S_{s,l} \cong 0 \dots [5]$$

2. Constant pressure cooling to  $T_e$  ("permanent" gases only)

$$\Delta S_g = \sum \frac{x_{ig}}{M_{ig}} \bar{C}_{pi} \ln \frac{T_e}{T_c} \dots [6]$$

$$\Delta S_g = \frac{x_{gc}}{M} \left( \bar{C}_{pg} \ln \frac{T_{be}}{T_c} - \frac{\Delta H_f}{T_{be}} \right) - \frac{x}{M} \frac{\Delta H_f}{T_f} + \frac{x}{M} \left( \bar{C}_{ps} \ln \frac{T_e}{T_f} + \frac{x_{lc}}{M} \bar{C}_{pl} \ln \frac{T_{be}}{T_c} + \frac{x}{M} \bar{C}_{pl} \ln \frac{T_f}{T_{be}} \right) \dots [7]$$

where  $x$  is weight fraction;  $T_{be}$ , the boiling point of the condensable species at  $P_e$ ; and  $T_f$ , the melting point which is taken independent of pressure. It has also been assumed in the above treatment that  $y_{lc} > 0$ .

There is another small entropy change due to the change in composition of the permanent gases after condensation. This may be calculated as follows

$$\begin{aligned} \text{Before condensation: } p^{p_0}_{ige} &= y_{ige} P_e \\ \text{After condensation: } p_{ige} &= y'_{ige} P_e \end{aligned} \dots [8]$$

where  $p$  is partial pressure, the superscript zero is the (fictitious) state at the end of the isothermal expansion and the prime denotes the composition with respect to the gas phase only. The entropy change of the permanent gases is then given by

$$\Delta S_g = \sum \left( \frac{x_i}{M_i} R \ln \frac{p^{p_0}_{ige}}{P_{ige}} \right) = \sum \left( \frac{x_i}{M_i} R \ln \frac{y_{ige}}{y'_{ige}} \right) \dots [9]$$

Inspection of cases of interest shows that one can select a mean value of the heat capacity for solid, liquid and gaseous condensable species which is at least as good as most of the other parameters. After equating the over-all entropy change to zero, solving for  $T_e$  and substituting in Equation [2], the final result for  $I_{sp}$  is

$$I_{sp} = \frac{1}{g} \left\{ \frac{2}{\bar{M}} \left[ \bar{C}_p T_c \left( 1 - \left( \frac{P_e}{P_c} \right)^{R(1-y_{lc})/\bar{C}_p} \times \exp \left( \frac{y_{gc} \Delta H_v}{\bar{C}_p T_{be}} + \frac{y \Delta H_f}{\bar{C}_p T_f} - \frac{R}{\bar{C}_p} (1 - y_{gc}) \ln (1 - y_{gc}) \right) \right) \right. \right. \\ \left. \left. + y_{gc} \Delta h_v + \Delta H_f \right] \right\}^{1/2} \dots [10]$$

The effect of condensable gases ( $y_{gc}$ ) is seen to be twofold: An increase in enthalpy (impulse) due to the presumed recovery of the phase change heats and a decrease due to the (exp) term and consequent increase of  $T_e$ . An attempt was made to maximize [10] with respect to  $y_{gc}$  for typical values of the other parameters and  $y_{gc} \gg y_{ge}$ . The optimum value of  $y_{gc}$  was greater than  $y$ , which means, of course, that the maximum of  $y_{gc}$  lies close to  $y$  where  $y_{gc}$  is not negligible; i.e.,  $T_e$  is close  $T_c$ . In other words, the impulse increases with increasing amounts of condensable gas up to the point where the temperature does not fall enough for condensation, the impulse then decreasing rapidly as  $T_e \rightarrow T_{be}$ .

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<sup>2</sup> Numbers in parentheses indicate References at end of paper.

If there are no phase changes on expansion, Equation [10] reduces to

$$I_{sp} = \frac{1}{g} \sqrt{\frac{2\bar{C}_p T_c}{\bar{M}}} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{R(1-\gamma)/\bar{C}_p} \right] \dots [11]$$

Glassner and Winternitz (1) have considered expansion with no condensation but have included variable heat capacities and the ratio of the solid velocity to the gas velocity. A trial and error process involving  $T_c$  is necessary to solve their equation. With constant heat capacities and velocity equilibrium, their expression reduces to Equation [11].

An expression similar to [11] frequently quoted (2) in both theoretical and experimental impulse work is

$$I_{sp} = \frac{1}{g} \sqrt{\frac{\gamma}{\gamma-1} \cdot 2n_0 R T_c} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma-1}{\gamma}} \right] \dots [12]$$

where  $n_0$  is the number of moles of gas per unit mass of propellant. Equations [11, 12] are identical provided that the heat capacity ratio  $\gamma$  is taken to be

$$\gamma \equiv \frac{C_p}{C_v} = \frac{\sum_g y_i C_{pi} + y_s C_{vs}}{\sum_{g,s} y_i C_{vi}} \dots [13]$$

and  $C_{vs} \approx C_{ps}$ . That is,  $\gamma$  should be calculated on the basis of all products in all phases. Maxwell, Dickenson and Caldin (3) have also derived an expression for expansion without phase change which is equivalent to [11, 12] except for a small factor which is probably a misprint. Their pressure ratio exponent in the present nomenclature is

$$\frac{R(1-y_{tc})}{\sum_g y_i \bar{C}_{pi} + \frac{y_s \bar{C}_{ps}}{1-y_{tc}}} \quad \text{rather than} \quad \frac{R(1-y_{tc})}{\sum_{g,s} y_i \bar{C}_{pi}} \dots [14]$$

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## Bending Dynamics of a Spinning Missile

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The effects of spin upon the bending of slender missiles with eccentric mass distribution are of great importance, particularly in the case of multistage missiles. The spin may be deliberately introduced for stabilization at launch or it may result from misalignment of the stabilizing fins. In either case the spin rate is time-dependent, and therefore the problem of determining these effects must be treated in a manner which is basically different from the usual "steady state" calculations. The purpose of this paper is to present a method for calculating the bending moments which result from a nonsteady spin rate and the eccentricity of the center of mass axis.

The differential equations for the deflections are solved by expansion of the deflection of the bending neutral axis in a series of Eigenfunctions of the body as a free beam. In this way, the variables are separated, and the time part of the solution is obtained using a digital computer.

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A sample problem is included showing the method of calculating the bending moments for a nonuniform missile with a time-varying spin rate and a specified eccentric mass distribution.

## Nomenclature

$a_n$	= vector defined in Equation [15]
$C_n$	= constant defined in Equation [11]
$D_n$	= $EI(x) \frac{d^2}{dx^2} \phi_n(x)$
$e(x)$	= eccentricity of center of mass from neutral axis
$EI(x)$	= flexural rigidity of the missile
$f_n(t)$	= time part of solution for $u_y(x, t)$
$F$	= external applied force
$g_n(t)$	= time part of solution for $u_z(x, t)$
$l$	= length of missile
$m(x)$	= mass per unit length
$M(x, t)$	= total internal bending moment
$M_z, M_y$	= component of $M$ in $z$ and $y$ directions
$P_y$	= force per unit length in $y$ direction
$P_z$	= force per unit length in $z$ direction
$T$	= external applied torque
$u(x, t)$	= displacement of neutral axis from $x$ axis
$u_y(x, t)$	= component of $u$ in $y$ direction
$u_z(x, t)$	= component of $u$ in $z$ direction
$\omega_n$	= $n$ th bending natural mode frequency
$\beta(t)$	= angular displacement of a line connecting center of mass with neutral axis
$\phi_n(x)$	= $n$ th Eigenfunction of missile in bending
$\Omega(t)$	= spin rate of missile

## Introduction

AS BALLISTIC missiles become more slender and include more stages, the problem of alignment is aggravated, especially when spin is introduced during the flight. The problem, then, is to determine the bending moments induced in the structure so that reasonable tolerances can be set on the location of the center of mass axis and on the rate of spin. If the bending moments which will actually occur in missile flight are to be found, it is necessary to consider the case where the spin rate is not constant but varies with time. Since the structure of the missile is usually a body of revolution, it can be assumed to be a circular bar with variable bending stiffness and variable mass density. The boundary conditions are that the ends are "free." Furthermore, the bending neutral axis can be assumed to be initially a straight line with the center of mass axis deviating from that line.

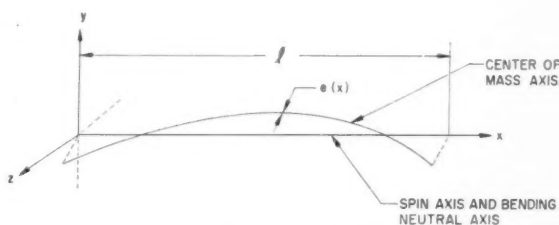


Fig. 1 Coordinate system

Since the bending of the missile results from the fact that the center of mass axis is not coincident with the spin axis, the eccentricity of the center of mass axis must be determined. But first it is necessary to locate the spin axis with reference to the body. Fig. 1 shows the center of mass axis for a rigid spinning body that is free from restraints. In this figure, the  $x$ ,  $y$ , and  $z$  axes are fixed in space and therefore do not rotate with the body. The  $x$ -axis represents the bending neutral axis as well as the spin axis. In locating the spin axis, it is assumed in this paper that no pitch or translation is introduced, since the pitch and translational inertia of slender bodies is relatively high. Then  $e(x)$ , the eccentricity of the center of mass with reference to the neutral

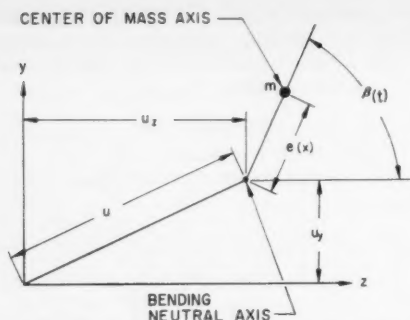


Fig. 2 Orientation of spin axis and fixed coordinate system

axis, must satisfy the conditions that linear and angular momentum must be conserved (1),<sup>2</sup> that is

$$\left. \begin{aligned} \int_0^l m(x)e(x) dx &= 0 \\ \int_0^l x \cdot m(x)e(x) dx &= 0 \end{aligned} \right\} \dots \dots \dots [1]$$

In most cases the function  $e(x)$ , and therefore the location of the spin axis, is adequately defined by these equations; the only error involved arises from neglecting the rigid body pitch and translation moments, both of which are usually negligible in problems of this sort.

Of course, the spin axis could also be located with reference to the center of mass by a consideration of the geometry of the body if the manner in which the spin is introduced is known. This approach, which would require a consideration of rigid body motions, is not used here; however, an explanation of how these motions can be included in the calculations is given later in this paper.

#### Discussion

##### Boundary Condition

Regardless of the method of analysis employed, since the ends of the body are free, the bending deflections must satisfy the boundary conditions

$$\left. \begin{aligned} \int_0^l m(x)u_s(x, t) dx &= 0 \\ \int_0^l x \cdot m(x)u_s(x, t) dx &= 0 \\ \int_0^l m(x)u_y(x, t) dx &= 0 \\ \int_0^l x \cdot m(x)u_y(x, t) dx &= 0 \end{aligned} \right\} \dots \dots \dots [2]$$

##### Derivation of Differential Equations

If a cut is made parallel to the  $y$ - $z$  plane, the beam will appear schematically (2) as shown in Fig. 2 where

$$\beta(t) = \int_0^t \Omega(t_1) dt_1 \dots \dots \dots [3]$$

The coordinates of the center of mass at this point are

$$\left. \begin{aligned} z &= u_s + e(x) \cos \beta(t) \\ y &= u_y + e(x) \sin \beta(t) \end{aligned} \right\} \dots \dots \dots [4]$$

Since the  $y$  and  $z$  axes are fixed in space, the equations of motion, found by equating the inertia load per unit length to the applied force per unit length, are

$$\left. \begin{aligned} P_z &= m(x) \frac{\partial^2 z}{\partial t^2} \\ P_y &= m(x) \frac{\partial^2 y}{\partial t^2} \end{aligned} \right\} \dots \dots \dots [5]$$

Since only bending forces are acting (2), Equations [5] become

<sup>2</sup> Numbers in parentheses indicate References at end of paper.

$$\left. \begin{aligned} -\frac{\partial^2}{\partial x^2} \left[ EI(x) \frac{\partial^2 u_z}{\partial x^2} \right] &= m(x) \left[ \frac{\partial^2 u_z}{\partial t^2} + e(x) \frac{\partial^2}{\partial t^2} \cos \beta(t) \right] \\ -\frac{\partial^2}{\partial x^2} \left[ EI(x) \frac{\partial^2 u_y}{\partial x^2} \right] &= m(x) \left[ \frac{\partial^2 u_y}{\partial t^2} + e(x) \frac{\partial^2}{\partial t^2} \sin \beta(t) \right] \end{aligned} \right\} \dots [6]$$

which are the differential equations of motion.

##### Separation of Variables

To solve these equations for the bending deflections, we assume a solution of the form (3)

$$\left. \begin{aligned} u_s(x, t) &= \sum_{n=1}^{\infty} g_n(t) \phi_n(x) \\ u_y(x, t) &= \sum_{n=1}^{\infty} f_n(t) \phi_n(x) \end{aligned} \right\} \dots \dots \dots [7]$$

where  $\phi_n(x)$ , the  $n$ th Eigenfunction of the body in bending, satisfies the subsidiary equation

$$\frac{d^2}{dx^2} D_n(x) = m(x) \omega_n^2 \phi_n(x) \dots \dots \dots [8]$$

and the boundary conditions, Equations [2].

Substituting Equations [7] in Equations [6] and making use of Equation [8] yields

$$\left. \begin{aligned} m(x) \sum_{n=1}^{\infty} \phi_n(x) \frac{d^2}{dt^2} g_n(t) + m(x)e(x) \frac{d^2}{dt^2} \cos \beta(t) &= \\ - \sum_{n=1}^{\infty} g_n(t) m(x) \omega_n^2 \phi_n(x), \dots, \end{aligned} \right\} [9]$$

Multiplying by  $\phi_m(x) dx$  and integrating from zero to  $l$ , the terms involving products  $\phi_m \phi_n$  drop out (where  $m \neq n$ ) because the modes are orthogonal. Then Equations [9] become

$$\left. \begin{aligned} \frac{d^2}{dt^2} g_n(t) + \omega_n^2 g_n(t) &= -C_n \frac{d^2}{dt^2} \cos \beta(t) \\ \frac{d^2}{dt^2} f_n(t) + \omega_n^2 f_n(t) &= -C_n \frac{d^2}{dt^2} \sin \beta(t) \end{aligned} \right\} \dots \dots [10]$$

$$n = 1, 2, 3$$

where

$$C_n = \frac{\int_0^l m(x)e(x)\phi_n(x) dx}{\int_0^l m(x)\phi_n^2(x) dx} \dots \dots \dots [11]$$

Equations [10] can be rewritten by performing the differentiation indicated on the right side

$$\left. \begin{aligned} \frac{d^2}{dt^2} g_n(t) + \omega_n^2 g_n(t) &= C_n \left\{ [\Omega(t)]^2 \cos \beta(t) + \left[ \frac{d\Omega(t)}{dt} \right] \sin \beta(t) \right\} \\ \frac{d^2}{dt^2} f_n(t) + \omega_n^2 f_n(t) &= C_n \left\{ [\Omega(t)]^2 \sin \beta(t) - \left[ \frac{d\Omega(t)}{dt} \right] \cos \beta(t) \right\} \end{aligned} \right\} \dots [12]$$

$$n = 1, 2, 3$$

The solution of these equations for  $g_n(t)$  and  $f_n(t)$  will allow evaluation of the bending deflections.

##### Bending Moment Equations

Since the rigid body motions are not considered at this point, the bending moments are given by

$$\left. \begin{aligned} M_s(x, t) &= \sum_{n=1}^{\infty} g_n(t) D_n(x) \\ M_y(x, t) &= \sum_{n=1}^{\infty} f_n(t) D_n(x) \end{aligned} \right\} \dots \dots \dots [13]$$

By vector addition, the total moment is

$$M(x, t) = \sum_{n=1}^{\infty} a_n D_n \dots \dots \dots [14]$$

where

$$a_n = \sqrt{f_n^2 + g_n^2} \dots \dots \dots [15]$$

From an examination of Equations [12, 13], it can be seen that an important consideration is the nature of the constant  $C_n$ , defined in [11]. The bending deflections and moments

Table 1 Normal mode computation

Station, in.	$EI$ , lb-in. <sup>2</sup> $\times 10^{-6}$	$m$ , lb/in.	$\phi_{1i}$ , in.	$\phi_{2i}$ , in.	$D_{1i}$ , lb-in.	$D_{2i}$ , lb-in.
0	400	3.7	-6.383	-2.135	0	0
48	625	3.7	-3.705	-0.094	-70,330	-180,500
96	625	3.7	-1.338	+1.171	-235,200	-415,600
144	2,000	7.5	+0.172	+1.000	-435,500	-351,300
192	1,900	11.2	+0.996	+0.278	-621,400	199,200
240	8,600	10.0	+1.000	-0.155	-734,700	1,004,700
288	37,000	42.0	+0.650	-0.215	-729,400	1,557,900
336	37,000	42.0	+0.246	-0.167	-543,721	1,470,200
384	37,000	42.0	-0.193	-0.022	-269,700	879,300
432	35,000	40.0	-0.652	+0.179	-60,900	221,329
480	20,000	18.0	-1.115	+0.400	0	0

$$\omega_1 = 35.5 \text{ rad/sec} \quad \omega_2 = 114.4 \text{ rad/sec}$$

are linear functions of  $C_n$ . Actually,  $C_n$  is the  $n$ th constant in a normal mode expansion for the weighted eccentricity function  $m(x) \cdot e(x)$ , indicating that the bending moments are functions of the shape as well as the amplitude of the weighted eccentricity.

As mentioned earlier, this procedure can be extended to include rigid body motions if desired. This can be done by including a term of the form  $(a + bx)$  in Equation [7] and by considering  $\omega_n$  equal to zero in Equation [10] for both the rigid body translation and pitch implied by this term.

#### Sample Calculations

The procedure described in this paper for calculating the bending moments is illustrated in the following example.<sup>3</sup>

For this problem, the shape of the center of mass axis was assumed, and from it, the eccentricity function was determined from Equations [1]. The normal modes,  $\phi_n$ , were then computed by the Stodola method using the  $EI$  and  $m$  distributions given in Table 1 for the body. The results of these computations are also shown in Table 1. The constant  $C_n$  was then computed from Equation [11]. Since the deflections and moments are linear functions of  $C_n$ , Equations [12] were solved for  $g_n/C_n$  and  $f_n/C_n$  (for  $n = 1$  and 2) based on the spin rate shown in Fig. 3. The amplitude and phase of  $a_1/C_1$  and  $a_2/C_2$  were computed using Equation [15]. The amplitudes are shown in Fig. 4. For this example, the bending moments were computed at  $t = 0.75$  sec, at which time  $a_1$  and  $a_2$  are in phase. The value of  $a_1/C_1$  is 0.953 and  $a_2/C_2$  is 0.0405 at this time. The bending moment was then calculated using Equation [14] and the results are shown in Table 2.

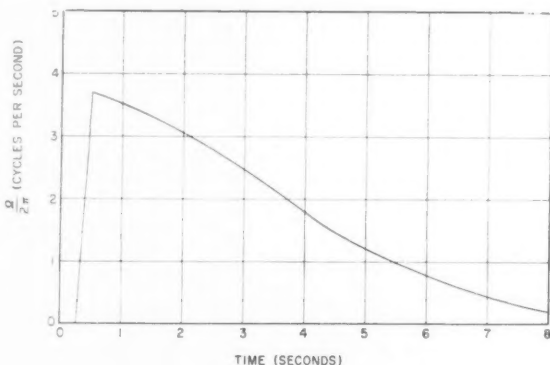
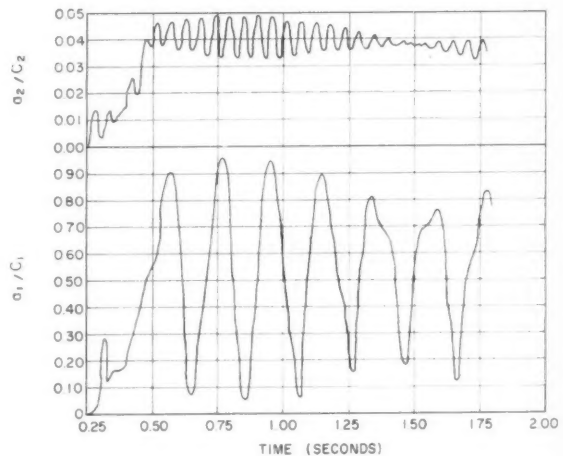
Fig. 3  $\Omega/2\pi$  vs. timeFig. 4  $a_1/C_1$  and  $a_2/C_2$  vs. time

Table 2 Bending moment computation

Station, in.	$e(x)$ , in.	$M(x)_t = 0.75$ in., lb-in.	$C_n$
0	-0.1742	0	
48	-0.0704	-1443	$C_1 = 0.00509$
96	+0.0330	-3680	
144	+0.1869	-4259	$C_2 = 0.1508$
192	+0.1737	-1797	
240	+0.0883	+2575	
298	-0.0662	+5981	$a_n/C_n$
336	-0.0315	+6389	
384	+0.0318	+4064	$a_1/C_1 = 0.953$
432	+0.0379	+1057	
480	+0.0725	0	$a_2/C_2 = 0.0405$

The moments used in this sample calculation are very small because the eccentricity chosen is unrealistically small for a body 480 in. long. If  $e(x)$  were chosen equal to  $0.1\phi(x)$ , which corresponds to a displacement of 0.64 in. at Station 0, then for the spin rate shown in Fig. 3, the maximum bending moment would be

$$M = 0.1 \times 0.953 \times D_1$$

that is, a maximum moment of 70,000 lb-in.; a moment of this magnitude could be an important consideration in the design of interstage connections.

The amplitude plots shown in Fig. 4 indicate that there is amplitude modulation at the natural bending frequency, which is to be expected from inspection of Equations [12]. It is important to note that the character of the response

<sup>3</sup>The author is indebted to the Numerical Analysis Department of Lockheed Missile Systems Division for the numerical solution of Equations [12] for  $f_n(t)$  and  $g_n(t)$ , and the solution of Equation [8] for the normal modes.

amplitude plot is essentially the same for both modes except that the first mode exhibits much higher amplitude oscillations. This is explained by the fact that the spin rate is always much closer to the first bending frequency than it is to the second.

### Conclusions

The problem of calculating the transient loads and moments introduced in a missile structure by nonsteady spin has been solved by making use of the separation of variables method. The space part of the problem involves calculation of the normal bending modes. The time part involves a differential equation which requires the use of a digital computer for solution. Problems of this type can arise in multi-stage missiles due to misalignment between the several stages, and in some cases could govern the design of the interstage connections. In general, nonsteady effects must be accounted for to adequately determine the loads and moments in the structure in question.

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## Stagnation Point Heat Transfer in Dissociated Air Flow<sup>1</sup>

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### Nomenclature

- $h$  = enthalpy  
 $H$  = total enthalpy  $h + u^2/2$   
 $h_D$  = dissociation enthalpy, defined as dissociation energy per unit mass in the external flow  
 $k$  = coefficient of heat conduction  
 $L$  = Lewis number  $\rho D \bar{c}_p/k$ , where  $D$  is the diffusion coefficient for the air atom, air molecule mixture and  $\bar{c}_p$  is the weighted average of the perfect-gas specific heats of the components, each weighted with the mass fraction of the component  
 $q$  = heat transfer rate to surface  
 $T$  = absolute temperature  
 $u$  = velocity along surface  
 $du_e/dx$  = velocity gradient along the surface in the external flow  
 $\mu$  = absolute viscosity  
 $\rho$  = mass density  
 $\sigma$  = Prandtl number  $\bar{c}_p \mu/k$ ; see  $L$  for definition of  $\bar{c}_p$

### Subscripts

- $e$  = external flow  
 $s$  = stagnation point  
 $w$  = wall  
 $\infty$  = free stream

### Superscripts

- $E$  = dissociation equilibrium flow  
 $F$  = chemically frozen flow  
 $*$  = evaluated at reference enthalpy

### Introduction

THE purpose of this note is to discuss stagnation point heat transfer in dissociated air flow and, in particular,

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to present the results of a recent theoretical investigation made at the AVCO Research Laboratory by J. A. Fay and F. R. Riddell.

The problem of obtaining the heat transfer rate may be considered in two parts. One is the determination of the properties of the gas, such as density, viscosity, Prandtl number, Lewis number. For high temperature dissociating gases such as air, these properties are not known accurately because of uncertain knowledge of the interaction properties of the atoms and molecules which make up the gas, and because of the chemical reactions which may take place. The other part of the problem is the solution of the boundary layer equations themselves for any given set of gas properties. Of course, any one numerical solution will be exactly valid only for the particular set of gas properties assumed. However, if the solution can be expressed in terms of the gas properties themselves, then we may expect it to be valid even for properties differing somewhat from the ones actually used to obtain the solution. That is, the solution will represent a relation between the heat transfer rate and the gas properties, whatever they may be. Approximate solutions in this form have previously been indicated by Subulkin (1)<sup>2</sup> and Lees (5); they will be described below. The exact AVCO solution presented in this note is also of this form.

### Previous Investigations

An early study of stagnation point heat transfer on a body of revolution was made by Subulkin (1). He considered only low speed flow and found for the heat transfer rate

$$q_s = 0.763 \sigma^{0.4} k (T_\infty - T_w) \sqrt{\left(\frac{du_e}{dx}\right)_s} \frac{\rho}{\mu} \dots \dots \dots [1]$$

Subulkin then suggested that this same relation might hold for a blunt-nosed body moving at supersonic speed, provided the free stream fluid properties were taken to be conditions behind the normal shock wave at the nose. This suggestion has been extrapolated to hypersonic speeds and dissociating flow (9) by introducing the total enthalpy instead of the temperature, so that Equation [1] becomes

$$q_s = 0.763 \sigma^{-0.5} (H_e - h_w) \sqrt{\rho_{es} \mu_{es} \left(\frac{du_e}{dx}\right)_s} \dots \dots \dots [2]$$

where the Prandtl number  $\sigma$  has been taken constant. This relates  $q_s$  to the values of specific fluid properties  $\sigma$ ,  $\rho_{es}$  and  $\mu_{es}$ . However, the relation is obtained by extrapolation from a low speed solution, rather than by solving the appropriate equations. Of course, the step from Equation [1] to Equation [2] is a pure guess, but subsequent exact solutions described below have shown it to be remarkably accurate in certain cases.

Mark (2) wrote the compressible boundary layer similarity equations for a blunt body stagnation point. However, he neglected the possibility of the heat transfer by diffusion being different from the heat transfer by conduction, thus confining himself to the case of Lewis number unity. He solved the differential equations using both perfect gas properties and the gas properties suggested by Romig and Dore (3) for air in dissociation equilibrium. The latter include variable Prandtl number and the thermodynamic properties of Krieger and White (4) which are now known to use an incorrect value of the dissociation energy of nitrogen. His results are expressed in the form

$$q_s = \frac{\lambda}{\sigma_w} (H_e - h_w) \frac{\rho_w \mu_w}{\sqrt{\rho_{es} \mu_{es}}} \sqrt{\left(\frac{du_e}{dx}\right)_s} \dots \dots \dots [3]$$

where  $\lambda$  is given numerically as a function of  $T_w/T_{es}$  and  $H_e$ . Though Equation [3] represents an actual numerical solution of the boundary layer equations, Mark does not relate  $\lambda$  to

<sup>2</sup>Numbers in parentheses indicate References at end of paper.

the transport properties, and therefore his results are difficult to apply to other fluid property variations, as well as being limited to Lewis number unity and equilibrium dissociation conditions.

Lees (5) has published a rather detailed paper on laminar heat transfer over blunt-nosed bodies in hypersonic flow, considering dissociation equilibrium with Lewis number unity, chemically frozen flow including nonunity Lewis number effects and a discussion of certain transport property values. However, he did not actually integrate the boundary layer differential equations. Instead, he inferred a solution from the constant property solutions of Cohen and Reshotko (8) by means of physical arguments about the nature of a highly cooled hypersonic boundary layer. For dissociation equilibrium with Lewis number  $L = 1$ , his result for the stagnation point is

$$q_s^E = 0.71 \sigma^{-2/3} H_e \sqrt{\rho_{es} \mu_{es}} \left( \frac{du_e}{dx} \right)_s \dots \dots \dots [4]$$

where the wall is highly cooled so  $h_w \ll H_e$ . In frozen flow he gives a result which may be put in the following approximate form for a diatomic gas

$$\frac{q_s^E}{(q_s^E/L=1)} = 1 + (L^{2/3} - 1) \frac{h_{pe}}{H_e} \dots \dots \dots [5]$$

where  $q_s^E(L=1)$  is from Equation [4]. Notice that Lees' Equation [4], and Equation [2] based on Subulkin's work, are almost identical. Both are essentially extrapolations from solutions for undissociated flow with constant  $\rho\mu$ . However, Lees used a good deal of clever physical reasoning to extrapolate, while the transition from Equation [1] to Equation [2] is a pure guess.

It should be pointed out that Lees' use of Cohen and Reshotko's value of the nondimensional enthalpy gradient as a solution to his Equations [9, 10] is actually wrong. The AVCO numerical integrations have shown that Cohen and Reshotko's value is too large by a factor  $(\rho_w \mu_w / \rho_{es} \mu_{es})^{0.9}$ , which may be as large as 4. However, when using this gradient in the heat transfer rate expression Lees states that  $\rho_w \mu_w$  must be replaced by  $\rho_{es} \mu_{es}$ , thus arriving at an approximately correct result. The present authors must say that they do not see the logic of this last step.

Recently Romig (6) has suggested a "reference enthalpy" method for finding  $q_s$  analogous to the well-known reference temperature methods of ordinary compressible flow. She begins with a form of Subulkin's equation

$$q_s = 0.763 \sigma^{-0.6} (H_e - h_w) \sqrt{\rho^* \mu^*} \left( \frac{du_e}{dx} \right)_s \dots \dots \dots [6]$$

where  $\rho^* \mu^*$  is to be evaluated at the reference enthalpy  $h^*$ . She proposes that  $h^*$  should be the average of  $H_e$  and  $h_w$ . She then specializes to the dissociation equilibrium properties of (4) and arrives at a form for  $q_s$  in terms of flight Mach number, nose radius and ambient pressure. To evaluate the reference enthalpy assumption she compares her result with Mark's for the same air properties and finds it agrees within  $\pm 5$  per cent. She has therefore, in effect, related Mark's numerical solutions to the fluid properties. However, the relation was made by way of the rather arbitrary assumption about where to evaluate  $\rho^* \mu^*$ , and only afterwards was it shown to agree with some actual numerical solutions. Notice also that for large wall cooling  $\rho\mu$  is quite flat in the outer half of the boundary layer so that  $\rho^* \mu^* \approx \rho_{es} \mu_{es}$  and Romig's correlation is very nearly the same as that obtained in Equation [2] from Subulkin's work.

#### AVCO Investigation

Concurrently with Lees' work a detailed investigation of stagnation point heat transfer in dissociating flow was

carried out at the AVCO Research Laboratory by J. A. Fay and F. R. Riddell. A more complete description of this work is being prepared for publication. Not only equilibrium and frozen boundary layers were considered, but also intermediate cases with finite recombination rates. Differences between heat transfer by conduction and by diffusion, corresponding to a Lewis number different from unity, were also considered. For the equilibrium case, the air properties of the NBS (7), which include the correct dissociation energy for nitrogen, were used. For frozen and finite-recombination-rate flow, air was taken to be a mixture of "air atoms" and "air molecules," with average properties determined by the relative concentration of oxygen and nitrogen atoms outside the boundary layer (that is, in the equilibrium state behind the normal shock in front of the body). Transport properties and recombination rates for high temperature air were estimated. In particular, for equilibrium, it was found that Sutherland's formula for viscosity was within 10 per cent of the estimates up to 9000 K. The Lewis and Prandtl numbers for atom-molecule mixtures were found to be substantially independent of temperature up to the same range, and were taken constant throughout the boundary layer in the calculations.

For all cases it was shown that the boundary layer equations for a stagnation point can be reduced exactly to ordinary differential equations by means of a similarity variable. With the fluid properties obtained as described above, the appropriate differential equations were integrated on an IBM 650 computer. The equations and transformations are similar to those given by Lees (5), but they were solved numerically rather than by inference from constant property solutions.

In the equilibrium case calculations were made for a range of wall temperatures from 300 to 3000 K and stagnation enthalpies  $H_e$  from 670 to 10,400 Btu/lb. This range corresponds to values of  $\rho_{es} \mu_{es} / \rho_w \mu_w$  from 0.17 to unity. Because of the uncertainty in the value of the (constant) Lewis number  $L$ , calculations were made for  $1.0 \leq L \leq 2.0$ . In most cases Prandtl number  $\sigma$  was taken to be 0.71. It turned out that the results of all these calculations could be correlated on the fluid property parameters  $\rho_{es} \mu_{es} / \rho_w \mu_w$  and  $L$  with an accuracy of about  $\pm 3$  per cent. Furthermore, a check showed that the low speed dependence on Prandtl number still appeared to hold in the present case. The result of the correlation based on these extensive numerical calculations is the following relation between heat transfer rate and fluid properties

$$q_s^E = 0.763 \sigma^{-0.6} (H_e - h_w) \left( \frac{\rho_w \mu_w}{\rho_{es} \mu_{es}} \right)^{0.1} \sqrt{\rho_{es} \mu_{es}} \left( \frac{du_e}{dx} \right)_s \times \left[ 1 + (L^{0.52} - 1) \frac{h_{pe}}{H_e} \right] \dots \dots [7]$$

The AVCO results thus show that the values of the fluid properties  $\rho\mu$  which appear in  $q_s^E$  are neither the wall nor the external flow values but a combination of both. Comparison with Subulkin's and Lees' formulas, Equations [2 and 4], shows that they differ for  $L = 1$  by a factor  $(\rho_w \mu_w / \rho_{es} \mu_{es})^{0.1}$ . Since the  $\rho\mu$  ratio may reach values of 4 or 5 for highly cooled boundary layers, it is seen that using the actual solution of the differential equations, Equation [7], may improve the answer by as much as 17 per cent for  $L = 1$ .

For frozen boundary layers, two cases must be distinguished. If the wall catalyzes recombination of the atoms, their dissociation energy is given up to the wall and atom diffusion to the wall can produce a large heat transfer when the free stream atom concentration is high. For this case, numerical computations show values of  $q_s$  almost identical with the equilibrium case if  $L = 1$ . This is not surprising since  $L = 1$  means that energy is transported equally well

by conduction and by atom diffusion, so it should not matter as far as heat transfer is concerned whether the atoms recombine in the boundary layer or pass through it and recombine on the wall. For  $L > 1$ , however,  $q_s^F$  for a frozen boundary layer is slightly greater than for an equilibrium one. A correlation similar to the one for  $q_s^E$  then leads to

$$q_s^F = 0.763 \sigma^{-0.6} (H_e - h_w) \left( \frac{\rho_w \mu_w}{\rho_{es} \mu_{es}} \right)^{-1} \sqrt{\rho_{es} \mu_{es} \left( \frac{du_e}{dx} \right)} \times \left[ 1 + (L^{0.63} - 1) \frac{h_{D_s}}{H_e} \right] \dots [8]$$

which again differs from Lees' Equation [5] by the one-tenth power of the  $\rho\mu$  ratio as well as slightly in the Lewis number dependence.

A few calculations were also done for the frozen boundary layer in the case where the wall is noncatalytic, thus preventing atom recombination. They show that in such a case, at high stagnation enthalpies, the heat transfer could be reduced by a factor of two or more. Accomplishing this saving depends, of course, both on being able to make a noncatalytic wall and on having a frozen boundary layer. The latter in turn depends on the ratio of recombination reaction time to the time it takes a particle to diffuse through the boundary layer. This ratio can be shown to be proportional to the square of the ambient density, as well as to the body nose radius (through the stagnation point velocity gradient). Thus the boundary layer would become frozen for flight at high enough altitudes, the more so for smaller bodies. A more quantitative statement cannot be made with any accuracy at this time because of the present uncertainty in recombination rates.

Some numerical calculations were also made for the finite-recombination rate case, which is intermediate between equilibrium and frozen, using the best estimates of recombination rates available. They confirmed the fact that for a catalytic wall the heat transfer rate is nearly the same as for both equilibrium and frozen cases, and thus nearly constant regardless of the chemical state of the boundary layer. The only change from  $q_s^F$  to  $q_s^E$  is a gradual transition from the Lewis number effect of Equation [7] to that of Equation [8]. For a noncatalytic wall they showed the drop off in  $q_s$  indicated above as the boundary layer went from an equilibrium to a frozen state.

Experiments in shock tubes have been made at the AVCO Research Laboratory by Peter H. Rose and collaborators in order to measure stagnation point heat transfer in equilibrium dissociated air flow. The stagnation temperature range covered was 3000 to 9000 K, which simulates flight velocities of 8000 to 27,000 fps at altitudes from 20,000 to 120,000 ft. These experiments are described in more detail in (10). The results are in agreement with [7] when the following air properties are used: The equilibrium thermodynamic properties given in (7), viscosity given by the Sutherland law, Lewis number 1.4 and Prandtl number 0.71.

### Conclusions

A detailed investigation has been made of stagnation point heat transfer rate in dissociated air flow. The appropriate differential equations, including the chemical effects, have been solved numerically for a set of assumed fluid properties, and the results correlated. This made it possible to express the heat transfer rate in terms of the fluid properties, Equations [7 and 8], so these expressions should be valid even for properties differing somewhat from those actually used in the calculations.

It was found that the chemical state of the boundary layer, if the wall catalyzed atom recombination, had only a slight effect on heat transfer rate. If the wall prevented atom recombination, the heat transfer rate could be reduced by as much as a factor of two, depending on how near the

boundary layer was to a frozen state. This in turn was found to depend on the size of the body, the ambient density and the atom recombination rate.

The results reported here are considered valid for stagnation enthalpies reached in flight at speeds up to the satellite velocity (approximately 26,000 fps).

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## Time-Temperature Relationships for Rupture of Metals in Combustion Atmospheres

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LARSON and Miller (1)<sup>2</sup> have used the Arrhenius equation to interpret the high temperature behavior of metals, either in creep or stress-rupture. An equation of the form

$$\frac{t}{t_0} = A e^{-Q/RT} \dots [1]$$

has been used, where

- $t$  = time to rupture, hours
- $Q$  = activation energy for the process
- $R$  = the gas constant
- $T$  = absolute temperature
- $A$  = a constant

Equation [1] is reduced to

$$T (\log A + \log t) = \frac{Q}{2.3R} = \text{const} \dots [2]$$

at constant stress, where  $\log A = C = 20$ , according to Larson and Miller.

While the rate equation implies that  $C$  is constant only when the stress is a constant, the Larson-Miller hypothesis

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<sup>1</sup> Metallurgist, Warhead Research Branch.

<sup>2</sup> Numbers in parentheses indicate References at end of paper.

maintains  $C$  is independent of stress and possibly of material.

If this is true, then  $Q$ , the activation energy of the process, must be stress dependent, since the equation contains no stress term, and all rupture times would be independent of stress, which is impossible. If the chemical rate process equation is used for the correlation of rupture data, then it must be admitted that either the pre-exponential constant or the activation energy, or both, must vary; both cannot be maintained as constants.

Evidence has been presented (2) showing a variation of the constant  $C$  with respect to applied stress. Further evidence of this is herein presented, although the conditions of testing include a possible gaseous diffusion process.

The method of testing has been previously described (3); that is, a statically loaded wire was heated in and by a flame of

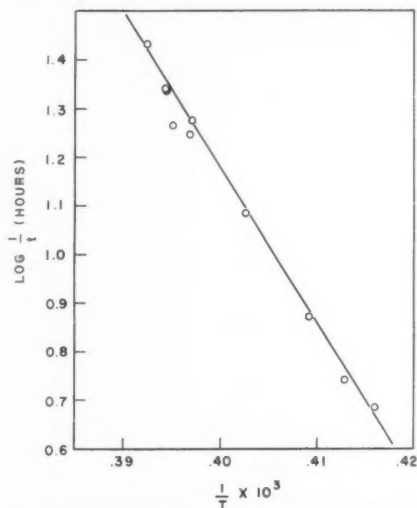


Fig. 1 Typical rate plot for Nichrome stressed at 6310 psi

burning propane gas. By maintaining a constant stress and varying the flame temperatures, various rupture times were obtained. Temperatures from 1900 to 2100 F could be achieved, and rupture times from 0.0083 hour (30 sec) to 1 hour were measured, while a series of ten separate stresses, from 5650 to 8860 psi, was employed. The resulting plots of  $\log 1/t$  against the reciprocal of the absolute temperature yield straight lines, the slopes of which determine the activation energies. Rankine temperatures have been used to conform to the original data of Larson and Miller (1). Fig. 1 is an example of such a plot, using Nichrome wire (80 nickel, 17 chromium) as a test material. Calculated values of the activation energies over ten stress levels are shown in Table 1.

Variations in the computed values of  $Q$  for chromel wire (60 nickel, 15 chromium and 22 iron) are even more marked (3), increasing from 35 kcal, below 5500 psi, to 48 kcal up to 9700 psi.

Table 1 Nichrome wire

Stress, psi	$Q$ , kcal
5650	79.2
5840	69.7
6125	76.1
6310	80.1
6600	78.8
6780	76.0
7250	83.4
7910	76.4
8380	76.1
8860	73.8

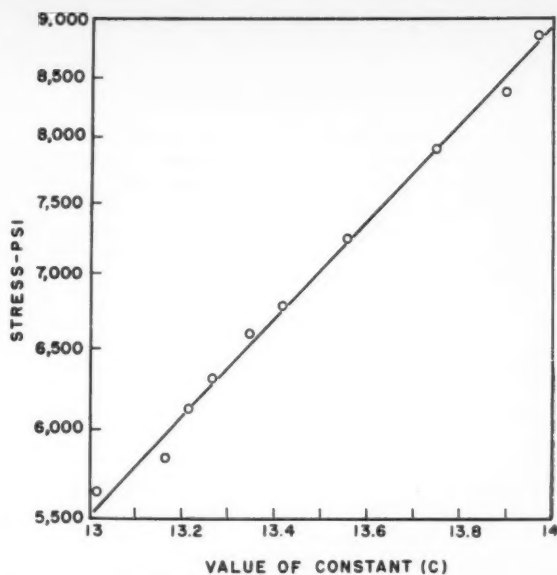


Fig. 2 Relationship between constant  $C$  and stress, assuming a constant activation energy; Nichrome wire

On the other hand, if the discrepancies in the values of  $Q$  in Table 1 are attributed to experimental error, and the activation energy is assumed to be constant, use of an average value of  $Q$  based on the data in Table 1 implies that  $C$  must vary.

If  $Q$  is maintained at 76.96 kcal, Fig. 2 shows the effect of stress on the value of the pre-exponential constant, which ranges between 13 and 14. This is in fairly good agreement with the data of Newhouse and Van Ullen (2) for austenitic steels.

It is not surprising that variations in the value of  $C$  exist, and when it is recalled that  $C$  is the logarithm of  $A$  in the rate equation, these variations are significant if extrapolation of data is desired. In the Arrhenius equation,  $A$  is actually slightly temperature dependent, and no simple physical significance has been established for processes similar to the one considered. In many first-order reactions (with the time measured in seconds),  $A$  has a value of  $10^{12}$  to  $10^{14}$ , so the present evaluation is not far out of line (4, 5).

Since this constant is also temperature dependent to some extent, the use of the high temperatures reported here, around 2000 F, is probably responsible for the lower value of the constant than that reported previously (1). It is concluded that with increasing temperatures of testing and application, and the shorter rupture times encountered, use of a simple material constant of 20 should be discouraged. It is believed that further research at these higher temperatures will be necessary to evaluate properly (a) the stress dependence of the activation energy, (b) the stress temperature relationships and (c) the true meaning and physical significance of the pre-exponential constant, if the chemical rate equation is to be used for accurate extrapolation of stress rupture data.

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# Effective Lift Drag Ratio With Jet Lift

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Downward jet deflection may be used either to augment or to replace aerodynamic lift. Optimum jet deflection angles are derived for both cases, and the resulting effective lift/drag ratios are plotted. Jet lift alone is shown to yield  $L/D_{\text{eff}}$  values ranging from 1.0 to about 4. For  $L/D_{\text{aero}} = 5$ , the use of optimum jet lift yields  $L/D_{\text{eff}}$  values ranging from 5.1 to about 7. In each case, the effect of jet lift increases as the jet thrust coefficient decreases.

## Nomenclature

$L_a$	= aerodynamic lift
$D_a$	= aerodynamic drag
$W$	= weight
$F_0$	= inlet momentum, $F_0 = \rho_0 A_0 V_0^2 = 2A_0 q_0$
$A_0$	= free stream capture area
$V_0$	= flight velocity
$F_j$	= exit stream thrust, $F_j = m_e V_e + A_e(P_e - P_0)$
$C_F$	= internal thrust coefficient based on $A_0$ , $C_F = (F_j - F_0)/(q_0 A_0)$
$C_D$	= external drag coefficient based on $A_0$ , $C_D = D_a/(q_0 A_0)$
$\phi$	= jet deflection angle measured from flight path
$L/D_a$	= aerodynamic lift/drag ratio
$L/D_{\text{eff}}$	= effective $L/D$ with jet deflection, $L/D_{\text{eff}} = (\text{aero lift} + \text{jet lift})/(\text{internal thrust})^*$

## Introduction

DOWNWARD deflection of a propulsive jet as a means for producing lift is utilized by aircraft operating at speeds throughout the flight spectrum. VTOL and STOL aircraft rely heavily upon jet lift, airplanes operating at moderate speeds almost neglect it, and, rather surprisingly, hypersonic vehicles may find it to be an important contribution to total

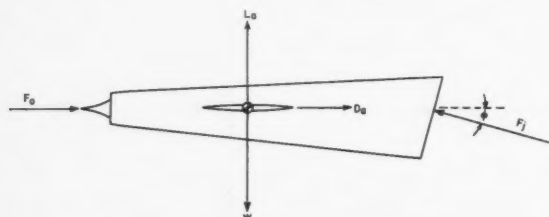


Fig. 1. Forces on a jet propelled aircraft

lift. This note presents an approach to optimized jet deflection which, while general in nature, is expressed in the nomenclature of supersonic and hypersonic aircraft performance.

## Combined Aerodynamic and Jet Lift

From the definitions given, expressions for optimum jet deflection angle and the resulting effective lift/drag ratio can be derived as follows

$$\frac{L}{D}_{\text{eff}} = \frac{\text{total lift}}{\text{internal thrust}} = \left( \frac{L_a + F_j \sin \phi}{D_a} \right) \left( \frac{D_a}{F_j - F_0} \right)$$

Since  $D_a = F_j \cos \phi - F_0$  for stabilized level flight with jet lift

$$\frac{L}{D}_{\text{eff}} = \frac{L}{D}_a \left( \frac{F_j \cos \phi - F_0}{F_j - F_0} \right) + \left( \frac{F_j \sin \phi}{F_j - F_0} \right)$$

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<sup>2</sup> Research Consultant.

\*  $L/D_{\text{eff}}$  relates the total weight-carrying ability of the aircraft to the net internal thrust (and thereby to the fuel consumption and engine size); hence it can be used directly in the Breguet range equation and it is applicable to problems where engine size is to be minimized for a given vehicle weight.

from the definitions of  $C_F$  and  $F_0$ ,  $F_j/F_0 = 1 + (C_F/2)$

$$\frac{L}{D}_{\text{eff}} = \frac{L}{D}_a \left( \frac{1 + (C_F/2) \cos \phi - 1}{C_F/2} \right) + \frac{(1 + (C_F/2) \sin \phi)}{C_F/2}$$

$$\frac{L}{D}_{\text{eff}} = \frac{L}{D}_a \left[ \left( \frac{2}{C_F} + 1 \right) \cos \phi - \frac{2}{C_F} \right] + \left( \frac{2}{C_F} + 1 \right) \sin \phi. [1]$$

Optimizing with respect to  $\phi$

$$\frac{d}{d\phi} \left( \frac{L}{D}_{\text{eff}} \right) = 0 = - \frac{L}{D}_a \left( \frac{2}{C_F} + 1 \right) \sin \phi + \left( \frac{2}{C_F} + 1 \right) \cos \phi$$

$$\tan \phi_{\text{opt}} = \frac{1}{L/D_a} \dots \dots \dots [2]$$

Returning to external drag

$$C_D = \frac{D_a}{q_0 A_0} = \frac{F_j \cos \phi - F_0}{q_0 A_0}$$

$$C_D = 2[(1 + (C_F/2) \cos \phi - 1)] \dots \dots \dots [3]$$

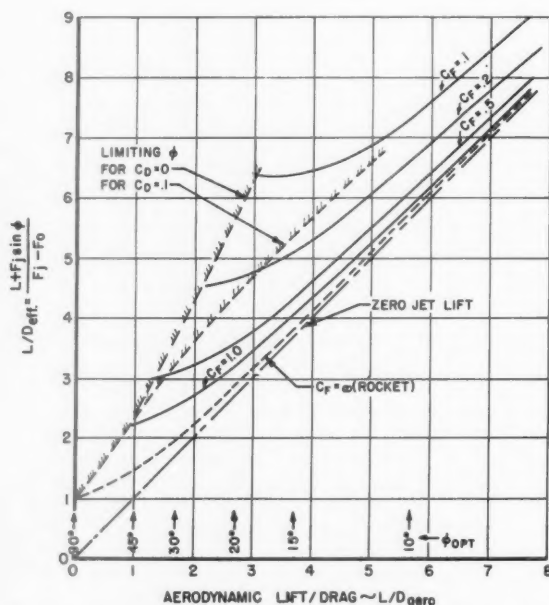


Fig. 2 Effective lift/drag ratio obtainable by combining aerodynamic lift with optimized jet lift

Fig. 2, which is based on Equations [1-3], presents an integrated picture of aerodynamic plus optimized jet lift. Effective  $L/D$  always increases with increasing aerodynamic  $L/D$ , so an optimum aerodynamic configuration is always desirable. Reducing  $C_F$  always leads to increased  $L/D_{\text{eff}}$ . However, a smaller  $C_F$  requires a larger engine and one of the following factors will ultimately limit the engine size:

1 Engine weight may increase to the point where it entirely counteracts the gains in  $L/D_{\text{eff}}$ .

2 Engine minimum fuel/air ratio or engine specific fuel consumption considerations may establish a minimum  $C_F$ .

3 Eventually, the external drag of the engine itself will prevent further reductions in  $C_F$ . In the extreme case where the configuration approaches a flying engine, the external drag coefficient may be of the order of 0.1; accordingly this value and  $C_D = 0$  (the absolute limit) are shown as illustrative boundaries on Fig. 2. Additional limiting  $C_D$  lines may be constructed by use of Equation [3].

### Jet Lift Alone

Even wingless vehicles are normally capable of useful aerodynamic lift/drag ratios, but there are certain cases where aerodynamic lift is not feasible. For such a vehicle, jet reaction alone may be used for support, and [1] reduces to

$$\left(\frac{L}{D}\right)_{\text{eff}} = \left(\frac{2}{C_F} + 1\right) \sin \phi$$

and Equation [3] can be rearranged to give

$$\cos \phi = \frac{(C_D/2) + 1}{(C_F/2) + 1}$$

by eliminating  $\phi$

$$\left(\frac{L}{D}\right)_{\text{eff}} = \frac{2}{C_F} \sqrt{C_F \left(1 + \frac{C_F}{4}\right) - C_D \left(1 + \frac{C_D}{4}\right)} \dots [4]$$

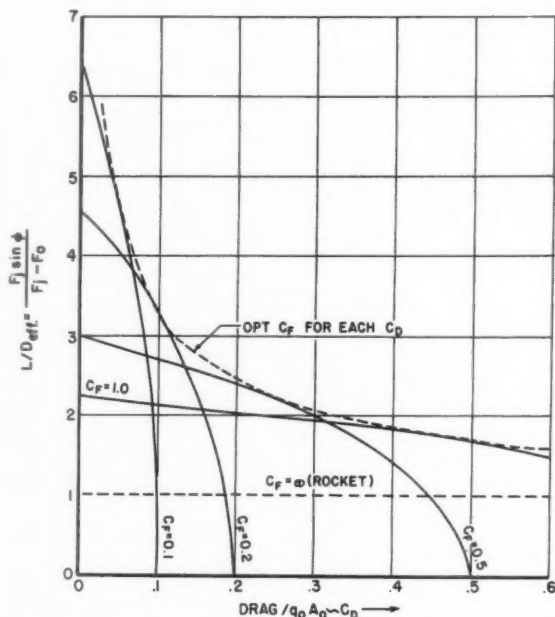


Fig. 3 Effective lift/drag ratio obtainable from jet lift alone

For given values of  $C_F$ , Equation [4] yields curves of  $(L/D)_{\text{eff}}$  as shown in Fig. 3.

If  $C_D$  is held constant, an optimum  $C_F$  may be found by differentiating Equation [4] with respect to  $C_F$  and setting the resulting equation equal to zero. This process yields

$$C_{F\text{opt}} = 2C_D + \frac{C_D^2}{2}$$

which leads to 
$$\cos \phi_{\text{opt}} = \frac{C_D}{2} + 1 \dots [5]$$

and for this case 
$$\left(\frac{L}{D}\right)_{\text{eff}} = \frac{1}{\sin \phi_{\text{opt}}} \dots [6]$$

The optimum  $C_F$  envelope curve of Fig. 3 has been generated through the use of Equations [5, 6].

Generally speaking, effective lift/drag ratios for jet lift only are disappointingly small. Where use of jet lift alone is necessary, it is important that both aerodynamic drag and thrust coefficient be minimized.

### Field Transportation of Concentrated Hydrogen Peroxide

(Continued from page 664)

vehicles is shown in Fig. 3. A study of this drawing will show how some of the design principles outlined in the preceding paragraph are applied in practice. The use of one four-way valve eliminates the possibility of trapping  $\text{H}_2\text{O}_2$  in a line due to improper setting of valves. With one setting of the four-way valve the system is ready for pumping from an outboard supply to the tank truck. With the other setting the contents of the truck tank may be discharged. A siphon break valve is included so that gross spillage may be prevented in case any of the piping is accidentally deranged.

The entire system is maintained dust tight to prevent entry of dirt. The vent line may be carried overboard to discharge the vented oxygen gas and prevent a buildup of an oxygen-rich atmosphere.

Certain details of a typical  $\text{H}_2\text{O}_2$  reservoir are shown in Fig. 4. There are no bottom connections that might leak. The baffles used to reduce slosh are kept small and are located so that a man can enter through the manhole to inspect all interior surfaces after the tank is fabricated and cleaned. A well for a temperature sensing device is provided. The sight port is used for observing the liquid level.

### Conclusion

Numerous trucks have been designed and are in present use in the field handling of  $\text{H}_2\text{O}_2$ . These units are fabricated mainly through commercially available components. Years of experience have demonstrated that concentrated hydrogen peroxide can be handled safely and easily in field applications.

### References

- 1 Bloom, Ralph, Jr., Davis, Noah, S., Jr., and Levine, (Continued on page 737)

## Research Engineers EXPLOSIVES

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## Fluorine Figures Big

**F**AR from new but equally distant from a practical operational status, the idea of a rocket engine using fluorine as an oxidizer refuses to die. It is, in fact, now more alive than ever, figuring in top-level discussions both public and private.

Theoretically, fluorine in combination with hydrogen or hydrogen-rich fuels makes an ideal propellant. Such systems combine high combustion temperatures with low molecular weights to form some of the highest specific impulses attainable in chemical propellants. They are generally hypergolic and smooth burning, and are composed of relatively abundant raw materials.

But pure hydrogen is an extremely difficult element to maintain in a liquid state, and fluorine, although less of a problem than hydrogen, is still a difficult element to keep liquid. Compared to liquid oxygen, liquid fluorine is also more expensive, more corrosive, more toxic and generally more of a handling hazard. Also, pumps suitable for liquid fluorine service are still not readily available.

**Yes, No and Maybe:** Fluorine's apparent advantages as an oxidizer even with fuels other than pure hydrogen are generally undisputed. Its drawbacks, however, are currently cause for a host of diverse opinions in the missile industry. That fluorine does have drawbacks is freely admitted even by the fluorine producers. The disagreement is about how serious they are—a matter of degree or comparison rather than demonstrable fact.

After extended consideration, two top rocket companies have tentatively decided that fluorine, compared to liquid oxygen, is still too expensive



NEW TRANSPORT and storage system moves fluorine to missile sites.

and hazardous for use in liquid rocket engines.

On the other hand, Rocketdyne's John Tormey concedes that the drawbacks are important but doesn't believe they are insurmountable. And, taking a more positive attitude, John Gall of Penn Salt Chemical Co. declares that the use of fluorine in liquid rocket propellants is "inevitable."

**Actions Count:** Biggest voice in actually determining fluorine's future, of course, is that of the government. Already heavily committed to liquid oxygen as an oxidizer, the government is apparently getting in even more deeply.

Although it has first option on this country's commercial high purity oxygen output of 32 billion cu ft/year, the government decided to build at least six captive liquid oxygen plants as well. And one of the plants, located at the nation's biggest missile test center on Patrick Air Force Base, which had a lax capacity of 10 tons/day, has since been expanded.

Still not satisfied, the government recently began a program to boost output by granting fast tax write-offs to civilian producers willing to supply an additional 4 billion cu ft/year of oxygen and nitrogen. This latest move, as interpreted in some quarters, is the result of a recently reached top echelon conviction that new oxidizers such as fluorine will not be ready by the time the first long-range ballistic missiles, now under development, become operational.

**Big Booster:** But this does not mean the government has lost interest in fluorine. For one thing, it has not withdrawn financial support from fluorine studies it is sponsoring that are now beginning to bear fruit.

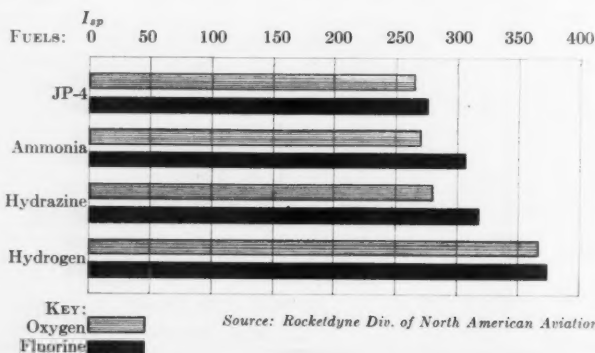
Rocketdyne's feasibility studies of a fluorine-using rocket engine, for example, begun under Air Force contract about six years ago, are continuing. And NACA's Lewis Flight Propulsion Laboratory, an early researcher of fluorine's rocket potential, is now studying the oxidizer's compatibility with various materials.

Meanwhile, of course, another government agency, the Atomic Energy Commission, as well as civilian fluorine producers, have been handling elemental fluorine for a number of years now and have collected a great deal of data on the subject.

More significant, perhaps, one producer, Allied Chemical, recently developed a method to produce and ship elemental fluorine commercially in liquid form. The tank transport system uses liquid nitrogen as a refrigerant and is designed to double as a stationary storage unit. The present unit, built by Air Products, is of a developmental size. But results to date are so favorable, reports Allied, that it is now building bigger transport units which will soon be in service.

**The Big Question:** What then is fluorine's future as an oxidizer in liquid

FLUORINE OUTREACHES OXYGEN  
Some Specific Impulse Values at 500 psia



Source: Rocketdyne Div. of North American Aviation, Inc.

propellant rockets? Even with its major drawbacks, most chemical and missile men feel that fluorine does definitely have a future as an lpr oxidizer. As one rocket researcher put it: "History has shown that when anything has strong advantages to offer, man will figure out some way to use it."

The question then becomes one of time and need. And right now, best estimates are that it will probably be at least two years before fluorine proves any threat to liquid oxygen and, perhaps, five years before it actually becomes the workhorse oxidizer of the rocket field.

### Introduction To Space Travel

A conference on space travel directed primarily at lay businessmen and bankers attracted some 400 interested persons to Birmingham, Ala., last month. It was believed to have been the first gathering of its kind, although a more general symposium had been held four years ago by a New York museum.

Seven speakers blanketed the field, covering fuels, metals and people for space craft, orbits and associated details for a trip to Mars, the earth satellite program, Russian missile progress, and the over-all need for basic research.

Sponsored by the Southern Research Institute, the two-day "Age of Space" gathering included, besides the speeches, a visit to Redstone Arsenal at Huntsville, home of the Army Ballistic Missile Agency on May 17. The tour featured rocket firing and missile handling demonstrations, said by base personnel to have been the most extensive yet for a nonclassified group.

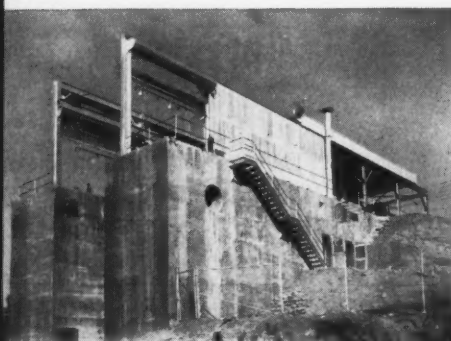
Dan A. Kimball, president of Aerojet-General, took the opportunity during his talk to announce that his company had statically fired on March 28 "the largest solid propellant rocket engine in the world." He added during questioning that two engines had, in fact, been successfully test fired at the Sacramento plant, and that the rockets were for a Navy ballistic missile (presumably the IRBM Polaris). The propellant used is a rubber-based compound, he revealed.

In his formal address, Mr. Kimball described one of Aerojet's most promising solid propellant materials as "a combination of fertilizer grade ammonium nitrate with synthetic rubber and an oil extender." Cost of the GR-S fuel is about six cents a pound, he said.

Maj. Gen. Dan C. Ogle, surgeon general of the Air Force, spoke on "People for Space Vehicles." In choosing psychologically stable persons for prolonged space travel, he speculated that "various steroids, hormones, and

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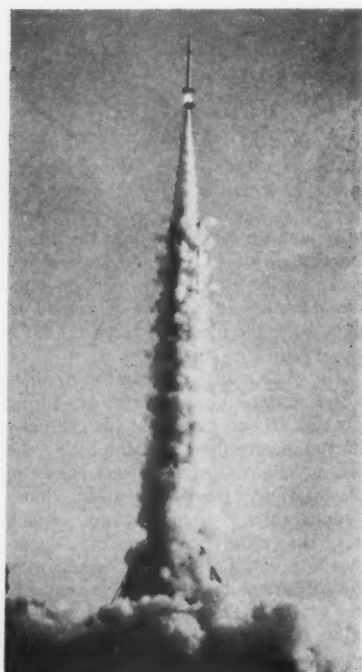
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Detroit 31, Michigan

other bio-chemical substances may be valuable indicators of latent stability, resistance, anxiety, heroism or fear. Human chemistry may determine our susceptibility to collapse or disease, and the electro-physical properties of the skin may tell scientists our capacity for continuing awareness."

Gen. Ogle also mentioned studies at the University of Texas on closed ecological systems for providing balanced removal of carbon dioxide and other wastes with input of oxygen, etc. It has been found, he said, "that about 2.3 kg of a certain green alga, the chlorella pyrenoidose, are sufficient to balance the gas metabolism of one man."

Other topics and speakers included "Metals for Space Travel," by F. L. LaQue of International Nickel; "The Earth Satellite Program," by John P. Hagen of the Naval Research Laboratory; "A Trip to Mars," by Ernst Stuhlinger of ABMA; "Russian Missile Technology," by Erik Bergaust of *Missiles and Rockets*; and "Military and Civilian Research," by C. C. Furnas of the University of Buffalo.



### Record Aerobee-Hi

Aerobee-Hi No. 41 fired at White Sands Proving Ground on April 30 set a world altitude record of 193 miles for single-stage booster rockets. The previous record was 164 miles. The missile, which landed 50 miles north of the launching site, achieved a speed of 4900 mph during its flight. Burnout time was 53 sec, altitude at burnout 147,000 ft and velocity at burnout 7200 fps.

JET PROPULSION

## MISSILE SYSTEMS PROPULSION

Weapon systems management activities at Lockheed's Palo Alto, Sunnyvale and Van Nuys organizations call for significant achievement in propulsion. Areas include design analysis, evaluation of test information and technical management of propulsion subcontractors. Inquiries are invited from those possessing a high order of systems ability and strong familiarity with solid and liquid propellant rockets and ramjets. Please address the Research and Development Staff, Sunnyvale 20, California.

*Here Propulsion Staff members discuss problems relating to accurate positioning of a vehicle in the upper atmosphere. Left to right: J. F. Houle, propellant feed systems analysis; B. Ellis, Propulsion Department manager; J. J. Donhan, control force generators.*

*Lockheed*

MISSILE SYSTEMS DIVISION

LOCKHEED AIRCRAFT CORPORATION

PALO ALTO • SUNNYVALE • VAN NUYS

CALIFORNIA



# Gasoline to Kerosene to 'Zip'—With Energy Calling the Signals

**M**ORE power in a smaller package, the common cry of the military since the air age began, has today assumed vital importance with the arrival of volume-limited aircraft and missiles.

To satisfy the energy cravings of their super- and hypersonic vehicles, the services are waiting impatiently for production quantities of the high energy chemical fuels from the "Zip" project. They are not marking time while waiting, however, but are instead searching for improved hydrocarbon fuels. In this quest, they seem to be shifting from gasoline to kerosene to provide the final push out of the petroleum era.

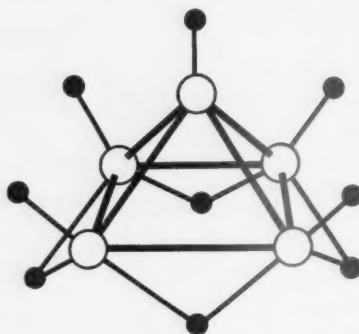
Military specifications for kerosenes have been issued for turbojets, ramjets, and rockets. The Navy is turning to kerosene for shipboard as well as land-based aircraft. Kerosene will fuel the first stage of the Vanguard vehicle.

But promising to outstrip even the best hydrocarbon fuels are the hydrogen-boron combinations, with more than half again as much combustion energy available to propel the vehicle.

With 50 per cent, or 9000, more Btu per lb from a fuel with a density like JP-4, presently the most popular jet fuel, the Air Force could, with a representative weapons system, "do any one of these or a combination of part of any of them:

"Fly at 60,000 ft instead of 40,000; go 5000 nautical miles instead of 3500; fly six hours instead of four; fly at Mach 3.0 instead of 2.5."

B. F. Wilkes, of the Aeronautics Division Directorate of Research and Development, Deputy Chief of Staff, Development HQ of the Air Force, gave these figures in calling for, in effect, fuels with more Btu per gallon. He predicted that the petroleum industry could expect turbojet fuel specifications calling for very low volatility and higher density, the latter held within tight limits. For ramjets and rockets, Mr. Wilkes saw several



Pentaborane ( $B_5H_9$ )

special-cut distillates with narrow API gravity and distillation ranges (30 to 40 deg, and 300 to 400 F, respectively).

**Versatile JP-4:** Returning to the present, two-thirds of all fuel burned by Air Force planes this year will be JP-4. Practically all of it goes into turbojets, although some rocket engines use it. The fuel has been described by oil companies as a wide-cut gasoline, effectively composed of 35 parts kerosene to 65 parts aviation reciprocating engine gasoline (AVGAS). Though the easiest fuel to make, it is not necessarily the cheapest. Indications are that it will hold its dominant position percentage-wise in consumption statistics for at least five years, as shown in Table 1.

Kerosene is the first cut from crude oil, boiling out of the distillation column at 300 to 500 F. In general it is a safe, cheap fuel, although high-grade kerosene now sells for about half cent a gallon more than JP-4 because of relatively limited production.

The turbojet (JP-designated) fuels in numerical order are:

- JP-1, a very expensive kerosene, the high cost due to a very low freeze point ( $-76^\circ F$ ) requirement. Demand is generally limited to subsonic jets of NATO countries, U. S. aircraft now being fed JP-4 and JP-5. The latest

specification, for JP-6, returns to many of the properties of JP-1.

- JP-2, an experimental fuel which never saw volume production.

- JP-3, in effect a high vapor pressure JP-4. Its use has been almost exclusively in land-based Navy jets; JP-5 is replacing it.

- JP-4, a wide-cut gas for planes up to Mach 1.5 capabilities.

- JP-5, a kerosene distinguished by a high flash point ( $140^\circ F$ ). Exclusively Navy in demand, with the elevated flash point dictated by shipboard safety rules. It fuels carrier jets capable of Mach 3.0 speeds.

- JP-6, a wide-cut kerosene featuring good thermal stability and clean burning characteristics, used in Air Force planes with Mach 1.5-2.0 capabilities. The thermal stability requirement has been necessitated by the increasing use of airborne fuel as the heat dump of the craft. Temperatures reached in the fuel tanks (in combination with pressures and dwelling times) are often enough to produce thermal cracking of the hydrocarbons, with attendant resin deposits in the plumbing.

For ramjet engines, RJ-1 is the only announced specification. Its diesel-like weight ( $7\frac{1}{2}$  lb per gal) gives it higher energy content than the JP family (average weight 5-6 lb per gal). High temperature stability and clean burning are other notable characteristics. The specification generally insists on smaller variations of specific properties than required of the turbojet fuels.

William S. Little Jr., a Shell Oil technical representative, considers the RJ-1 spec "a high quality approach to ramjet fuels." The military "wants a material not as easy to ignite as the JP fuels (but at the same time) wants as high an energy content per pound as possible, or high energy per gallon, because such craft are normally volume limited."

Finally, the only publicized rocket engine fuel, RP-1, is a lighter cut kerosene than RJ-1. It boasts few contaminating aromatics, with a high level of burnability.

**Enter B-H Combinations:** Successful development of the high energy boranes, the compounds of boron and hydrogen, represent the culmination of a decade of American effort on combustion studies and the use of the compounds in rocket motors, ramjets and air-breathing engines. Intensive studies began with liquid and solid boranes in 1952 with the Navy's "Zip" project. Production quantities are scheduled for next year.

Besides claiming for the borohydrides

**Table 1. Air Force Fuel Estimates**

(Millions of gallons per fiscal year)

	1957	1958	1959	1960
AVGAS	1350	1400	1400	1350
JP-4	3000	3800	4500	5000
JP-6	0.5	3	4	5
RJ-1	0.5	3	5	6
RP-1	1	5	10	11

Source: AIA.

Both are sealed...



### *A new series of Solenoid Valves —*

completely **hermetically sealed** and **qualification tested** — are now provided by Futurecraft for aircraft and missile use! These completely reliable and thoroughly tested and sealed valves are designed for all applications where high or low pressure helium, nitrogen, compressed air and other gases are used.

Look at the important advantages these qualified valves give you...

**Hermetically Sealed**—these valves are absolutely unaffected by moisture, sand, dust, fog, frost or ice under any conditions of use.

**Qualification Tested**—under MIL-E-5272A, MIL-S-4040A, NA 5 and the AIA Rocket Technical Committee requirements, Futurecraft Valves have passed the toughest testing specifications imaginable.

Types Available:	2-Way	} N.O. or N.C.
	3-Way	
	4-Way	
Direct or pilot operated Standard or miniature models		

#### **OPERATION**

Positive operation under all inlet pressures from 0 to 3000 psi

#### **LEAKAGE**

ZERO internal and external with helium at all pressures

#### **DUTY CYCLE**

Continuous duty (72 hr. test) @ 30 V at +165°F.

#### **RESPONSE**

60 milliseconds maximum to energize or de-energize under temperature extremes.

#### **VIBRATION**

Far exceeds requirements of MIL-E-5272A. Tested 32 hrs. under critical resonance @ 22G acceleration, 20 to 2000 cps.

#### **CYCLE LIFE**

In excess of 50,000 cycles @ 3000 psi.

#### **PROOF PRESSURE**

5,000 psi.

#### **BURST PRESSURE**

7,000 psi minimum.

#### **WEIGHTS**

As low as 8 oz.



# *Futurecraft*

**CORPORATION** 1717 North Chico Avenue • El Monte, California

*specializing in control of high pressure pneumatics and corrosive liquids for the missile industry*

*If you have a valve application involving control of high pressure gases, obtain the exclusive advantages of Futurecraft hermetically sealed and fully qualified valves. Futurecraft will gladly provide a complete qualification test report to authorized persons on request. Send for your copy today, and outline your requirements.*

## PRESSURE REGULATION

Far Beyond...



If you are now working with gases or liquids under extreme pressures—or if you plan engineering in the missile field far beyond pressures now considered normal—then you need the precise control and performance of Futurecraft Regulators. Here's why...

Typical operating characteristics cover regulation of inlet pressure as high as 4400 psi with outlet pressure down to 10 psi—with accuracy of plus or minus .5 psi.

High flow rates range to 12,000 scfm of helium under regulated pressure of 665 psi.

Operation to rated capacity is maintained under ambient temperature ranges from  $-260^{\circ}\text{F.}$  to  $+400^{\circ}\text{F.}$ —a vital factor for missile use.

Every regulator type—dome loading piston or diaphragm types...single or dual stage...with or without integral relief valve...programming regulators giving two different outlet pressures on electrical signal...liquid regulators—are Futurecraft engineered for the aircraft and guided missile industry.

*Whatever your problem in high pressure regulation and control of gases or liquids, bring it to Futurecraft. Our engineering staff will gladly work with you.*

**Futurecraft**  
**CORPORATION**

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### WRITE FOR FURTHER DATA

Futurecraft designs and manufactures for aircraft and guided missiles the following valve types: Solenoid Valves, Blade Valves, Propellant Valves, Pressure Relief Valves, Manual Control Valves, Pressure Regulators, Shuttle Valves, Check Valves, Line Valves and Filters, Explosive Valves, Butterfly Valves, Vent and Relief Valves, Burst Diaphragms, Quick Disconnect Couplings and Irradiated Polyethylene and Teflon "O" Rings.

*specializing in control of high pressure pneumatics and corrosive liquids for the missile industry*

*at your Fingertips*

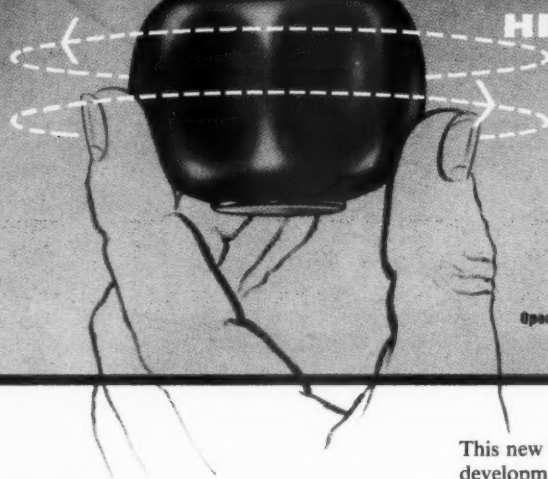
INSTANT PRESSURE REGULATION

from 0-3000 psi

WITH THE

*Futurecraft*

HI/LO HAND LOADER



GENERAL SPECIFICATIONS

Port Size:	1/4" tube per ANSI 10086
Pressures:	Operating - 0-3000 psi Proof - 6000 psi Burst - 7500 psi Minimum
Flow:	Equals .125" diameter sharp edge orifice
Operating Temperatures:	-65°F to +250°F
Weight:	2 1/2 pounds

Here is a highly sensitive, quick response pressure regulator for accurately controlling gases under high pressure. The Hi/Lo Hand Loader covers the widest possible range of downstream pressures, and once set does not allow "creep" or drift from established gauge readings. The unit is of the zero bleed, internal venting type, simple hand adjustment clockwise providing for pressure regulation, and hand adjustment counter-clockwise providing for pressure relief over the 0-3000 psi operating range.

This new Futurecraft valve is the result of four years' development work, spent to produce a wide range, precise unit for accurately testing rocket and pneumatic components. It is extremely simple, assuring long, trouble-free service. There is no diaphragm to break, no parts to require adjustment, no constant lubrication required. Porous bronze inlet and outlet filters furnished with the unit are easily accessible for cleaning, and a simple mounting bracket provides for any panel installation requirement. Typical uses include HELIUM, nitrogen and compressed air service in calibrating and testing, and as an adjustable relief valve.

Representatives'  
inquiries  
invited

*If you need convenient, accurate and sensitive regulation of high pressure gases, this new Futurecraft Hi/Lo Hand Loader can help.*

WRITE FOR FURTHER DATA

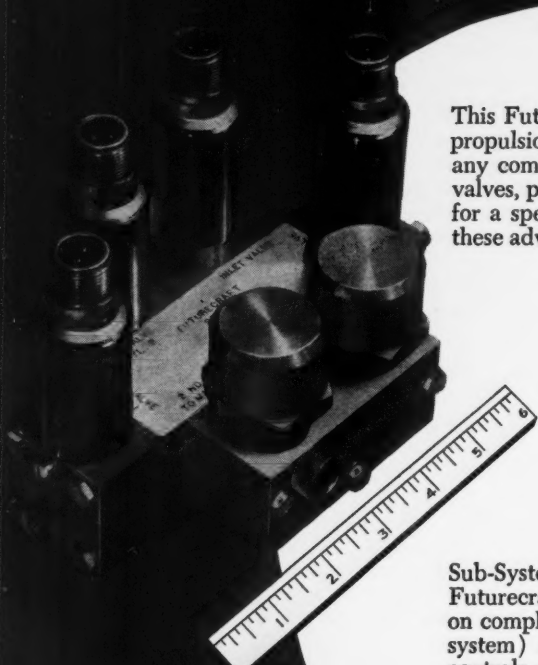
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# Futurecraft



This Futurecraft manifold system controls gas or fluid flow of propulsion system pressurizing in rockets and missiles. Using any combination of filters, regulators, master valves, solenoid valves, pressure releases — or other control components needed for a specific application — this system of packaging provides these advantages...

Up to  $\frac{1}{3}$  saving in weight over conventional panel type mounting

Space economy, achieved with extremely compact design and construction

Elimination of piping — and leakage or failure of connecting components

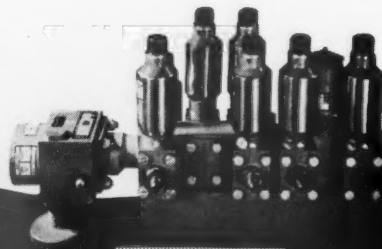
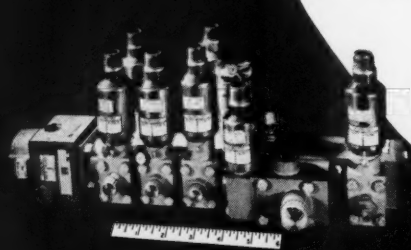
Simplification of check-out procedures, inspection, installation, replacement and procurement

Reduced cost through simplified system and standardization of parts

Sub-System Manifolding, designed, engineered and built by Futurecraft for leading missile manufacturers, has been proven on complex projects. Operating pressures to 4,500 psi (helium system) are handled with *zero* leakage. A wide variety of controls can be established in a single system to meet specific requirements. Each system is built, tested and checked out as a complete package.

If manifolding is *your* problem, find out how Futurecraft can help. Our engineering staff will gladly supply further details — or make a specific recommendation.

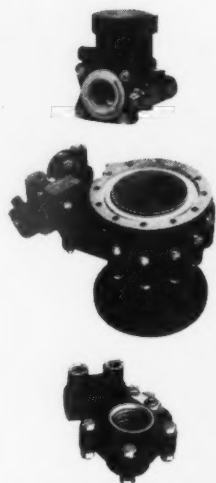
**Call — or write — today!**



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# The best-behaved propellants are *Futurecraft* controlled!



Futurecraft Propellant Valve used on the North American Aviation 60,000 pound thrust rocket engine which propels the Cook Electric Research Laboratories sled.



## General Specifications BLADE VALVES

SIZES: from 1/2" tube to 7" through port  
BODY: 356-T6 aluminum alloy or 61 ST  
PARTS: heat-treated 416 stainless steel  
347 stainless steel blades  
PACKING: Teflon or Kel "F" with Teflon  
inserted friction-free pressure pads

Putting a harness on rocket propellants is a mighty tough problem in the missile industry. But even fuming nitric acid is as meek as a kitten under the controlling hand of these Futurecraft Blade Valves!

Here are some of the important control features designed and built into these valves for your control problems:

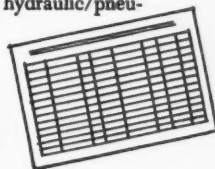
1. You get absolute minimum pressure drop and unobstructed, non-turbulent flow.
2. Extremely accurate and consistently controlled openings and closings are provided.
3. Valves can be adjusted for controlled oxidizer or fuel lead during rocket motor starting transient.
4. Built in by-passing simultaneous with cut off of the main flow, eliminates shock loads on closing.
5. Valves can be used as variable orifice control valves with proper slave system.

And to top it off, these Blade Valves have ZERO leakage, and are designed to handle operating pressures to 1000 psi and temperatures from  $-300^{\circ}\text{F}$  to  $+250^{\circ}\text{F}$ . They are proven in service for propellants and liquids such as liquid oxygen, RFNA, WFNA, aniline, hydrazine, ethylene oxide, propyl nitrate, JP-4, JP-5, hydrogen peroxide, air, nitrogen and water.

Blade Valves are only one of the many valve types engineered and built by Futurecraft for the guided missile and aircraft industries. Let us help you. Send your specifications for a prompt recommendation. Futurecraft specializes in development of hydraulic/pneumatic components, and your inquiry will receive quick attention.

## Send For Your Valve Selection Chart!

To help you with your engineering, write for this valuable selection chart. Valves made by Futurecraft are detailed as to size and type, actuating means, material, packing, weight, temperature range, operating pressures, flow characteristics and other helpful facts. It's yours for the asking—send for it today!



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*specializing in control of high pressure pneumatics and corrosive liquids for the missile industry*

To regulate, control, check or  
relieve gases and liquids —  
you will engineer better  
with valves by

*Futurecraft*

Single coil,  
3000 psi normally  
closed 3-way valve,  
pilot operated for hydraulic  
or pneumatic service.

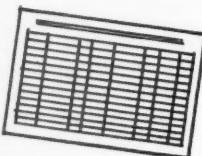
If precision operation of a gas or fluid system  
is your problem, you can achieve the  
ultimate in flow control with valves by Futurecraft!

Shown here, for example, are Futurecraft Solenoid Operated Valves — only a few of the scores of valve types designed and engineered by Futurecraft for the guided missile and aircraft industries. These components combine light weight, precision operation and finest workmanship with temperature, pressure and flow characteristics to meet the most exacting standards. They are giving outstanding performance and have solved a whole range of control problems in modern missile and aircraft flight techniques.

Send us your specifications. Futurecraft specializes in development and building of hydraulic/pneumatic components, and we will be glad to give our recommendation based on your requirements. Get in touch with Futurecraft — today!

Write for this Valve Selection Chart!

This valuable chart — yours for the asking — gives basic information on all Futurecraft Valve sizes and types, including engineering data on: actuating means, type of service, material, packing, weight, temperature range, flow and pressure characteristics and other facts to help you. Send for your Selection Chart today. No obligation, of course!



GENERAL SPECIFICATIONS—SOLENOID VALVES								
Description	Types	Sizes	Leakage Rate	Operating Pressures	Voltage	Operating Temperature	Duty	Proven in Service For
Single Coil Direct Operating Solenoid Valve	On-Off 2 Position, normally closed or open— 3-Way 2 Position, normally closed— 4-Way 3 Position with or without neutral and/or manual override	1/8", 1/4" tubing for D.C. Valves to 1/2" tubing for A.C. Valves	Zero	Variations to 4,000 psi —proof pressure 7,500 psi	18-30V. D.C. and 110V. A.C.	-65°F. to +165°F. with variations to +500°F.	Continuous	Air, Nitrogen, Helium, Oxygen and Hydraulic Oil
Single Coil Pilot Operated Solenoid Valve	On-Off 2 Position, normally closed or open— 3-Way 2 Position, normally closed— 4-Way 3 Position with or without neutral and/or manual override	To 1 1/4" tubing	Zero	Variations to 3,000 psi	18-30V. D.C.	-65°F. to +165°F. with variations to -300°F. and +500°F.	Continuous	Air, Nitrogen, Helium, Oxygen, Hydraulic Oil, R.F.N.A., W.F.N.A., Aniline, Hydrazine, Ethylene Oxide, Propyl Nitrate, Liquid Oxygen, J.P.-4, J.P.-5, Hydrogen Peroxide
Double Coil Direct Operating Solenoid Valve	On-Off 2 Position, normally closed or open	To 1/2" tubing	Zero	Variations to 3,000 psi	18-30V. D.C.	-65°F. to +165°F. with variations to -300°F. and +500°F.	Continuous	Air, Nitrogen, Helium, Oxygen, Hydraulic Oil, R.F.N.A., W.F.N.A., Aniline, Ethylene Oxide, Hydrazine, Propyl Nitrate



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**Table 2. Physical Properties of Hydrocarbon Fuels**

	Distillate Range, °F	Gravity, deg API	Freeze Point, °F, max	Flash Point, °F, min	General Description
JP-1	400-570	35 (min)	-76	110	Low freeze kerosene
JP-3	150-500	50-60	-76	NR	High vapor pressure JP-4
JP-4	200-550	45-57	-76	NR	Wide-cut gasoline
JP-5	350-550	36-48	-40	140	High flash kerosene
JP-6	250-550	37-50	-65	NR	Thermally stable kerosene
RJ-1	400-600	32.5-36.5	-40	190	Thermally stable, heavy kerosene
RP-1	380-525	42-45	-40	110	Pure, light cut kerosene

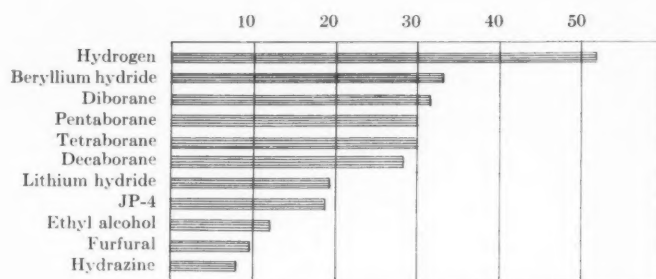
NR: No requirement.

60 per cent greater heat of combustion than JP-4, developers have mentioned increases of 15 to 35 per cent in the specific impulse rating. Further, one developer, Callery Chemical Co., says the compounds "can be used efficiently at altitudes where ordinary fuels will not burn."

Heats of combustion for the B-H combinations have been placed around 30,000 Btu per lb, in contrast to the 18,500 of hydrocarbon fuels. Hydrogen releases 52,000 Btu per lb, the highest value for any pure material, while boron gives up 25,000 Btu. In addition, boron is one of the best liquid carriers of hydrogen. A stabilizer such as carbon, added to make the combination less touchy, may reduce the final energy release to 28,000 Btu or so, but that's still a 50 per cent jump over petroleum fuels (see chart).

(These combustion figures are based on oxygen as the oxidizer. Use of fluorine instead of oxygen would give greater performance, according to one fluorine expert.)

Earl A. Weilmuenster of the Olin Mathieson Chemical Corp., a second developer of the high energy fuels, revealed last month that pentaborane



Btu/lb × 10³

Source: Olin Mathieson Chemical Corp.

#### Heat value of propellants

and decaborane are the liquid propellants getting the most attention (see Table 3). The usual route to these higher boranes is through diborane, with the simple application of heat to perform the conversion; "many" other ways, not cited in his paper, were also available, according to Mr. Weilmuenster.

Diborane itself is prepared by the addition of lithium hydride to boron trifluoride etherate. Lithium fluoride and ethyl ether are by-products of the reaction, but the diborane, being a gas, is conveniently removed from reaction medium. The stoichiometric equation given by the scientist was



Parenthetically, it may be noted that fear had been expressed late last year that removal of contaminating traces of hydrogen and nitrogen from the gaseous product could be a "key drawback" to the fuel's manufacture. Fractional distillation at low temperature, cited as the usual method, was very costly. Ralph K. Birdwhistell and co-workers at Michigan State University then came up with evidence that separation could be effected by diffusion through a plastic membrane, the diborane slipping through much faster at room temperatures than the other two gases. No mention of how useful the discovery has been to production was heard, however.

Olin Mathieson will produce two high energy fuels, designated HEF-2 and HEF-3, presumably the penta- and decaboranes, at a \$36-million plant for the Air Force and at a smaller unit for the Navy. Both facilities are being erected at Model City, New York. Also, the company's pilot plants in nearby Niagara Falls are turning out experimental quantities of the

**Table 3. Physical Properties of Boranes**

	Molecular Weight	Melting Point, °F	Boiling Point, °F	Specific Gravity
Diborane (B <sub>2</sub> H <sub>6</sub> )	27.7	-265	-134	0.43
Pentaborane (B <sub>5</sub> H <sub>9</sub> )	63.2	-52	140	0.61
Decaborane (B <sub>10</sub> H <sub>14</sub> )	122.3	211	415	0.94

Source: Olin Mathieson Chemical Corp.

fuels, it has been reported.

Callery, with its HiCal boron fuel for the Navy, broke ground in March on a \$38-million plant in Oklahoma, and last month announced the beginning of work on a \$4-million intermediates plant in Kansas.

In addition, American Potash and Chemical last month revealed that it has produced some decaborane. The company, one of three in the non-Communist world that produces boron, mentioned no production plans. Other boron suppliers, U. S. Borax and Chemical and Stauffer Chemical, are reported to be researching the field and producing intermediates, respectively. Development contracts for application of the fuels to jets have been let to a number of big aircraft companies, among them North American Aviation and Boeing Aircraft.

Some unique materials such as slurries of aluminum and magnesium have been investigated by the National Advisory Committee for Aeronautics. Another novel path, that of using the beta radiation from isotopes to speed the burning of a gaseous fuel, has been successfully tried by Stuart Churchill and his colleagues at the University of Michigan.

But it seems evident that the big push in propellants in the foreseeable future will be in the boron-hydrogen combinations where all desirable characteristics of a fuel can be found: High energy content per unit weight and per unit volume, easy handling and low cost.

### British Missile

Firebreak, an air-to-air missile with infrared homing, has been put into production by the British Royal Air Force for the Electric P.1. and the Gloster Javelin fighter aircraft. Charged with turning out the weapon is the developer, de Havilland Propellers Ltd. of Hatfield, England.



De Havilland Firebreak homes on target.

## Crystalline 'Glass' Noses into Missile Field

A new family of crystalline materials, made from glass, but extremely hard and fine-grained, was revealed last month by Corning Glass Works. It has already been centrifugally cast into missile radomes.

The materials, named Pyrocerams, are reportedly harder than high carbon steel, lighter than aluminum, and up to nine times stronger than plate glass. More specifically, they are said to exhibit high mechanical strength, excellent electrical insulating properties, superior thermal shock resistance, high deformation temperatures and good chemical resistance.

Discoverers see the ceramic-like substances finding wide use in supersonic aircraft, jet engine components, the chemical and oil-refining industries (as piping), and the home.

"Certain types of Pyroceram keep their strength at temperatures as high as 1300 F," Corning officials stated. They hailed the discovery, by S. Donald Stookey of their fundamental research labs, as one of the greatest technological advances in glass research since heat-resistant borosilicate glass was found early in this century.

Softening temperatures range up to 2460 F for the four types of Pyroceram that have been fully investigated. These, with a specific gravity variation of 2.40 to 2.62, were chosen from about 400 that were experimentally melted. About a thousand materials have been conceived.

**Heat Expansion Variable:** Tailor-made coefficients of thermal expansion, from slightly negative (contraction) to high enough to match heavy metals ( $200 \times 10^{-7}/\text{degree C}$ ), can be achieved, suggesting bonding of Pyroceram to metal for high temperature applications. Pyroceram by itself can show flexural strengths as high as 60,000 psi and strength-to-weight ratios greater than titanium or stainless steel, the company claims.

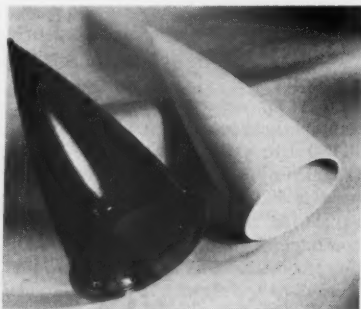
In the manufacturing process, one or more nucleating agents are compounded into the special glass batch. The fluid melt is formed into the desired shape by conventional pressing, drawing, blowing, rolling or centrifugal casting glass techniques. (Certain Pyrocerams can even be investment cast for complex, close tolerance parts, it is said.)

Still glassy in appearance (see photo), the piece is first cooled, then given a series of heat treatments that cause the nucleating agents to form billions of submicroscopic crystallites per cubic millimeter. These crystals may vary in size from a few hundred

to ten thousand Angstrom units, with each crystallite acting as a center of growth as the heat continues in the furnace. Machining is possible after the final cooling.

The finished piece can be transparent or opaque. Because of its impermeability to gases (as good as the best glasses), it is pictured as a construction material for high vacuum apparatus. Further, the material has the capability of insulating within an electric field and yet not becoming heated itself. And, as a polycrystalline oxide, it does not oxidize.

**More Expensive Than Glass:** Only pilot plant volumes have been produced by Corning so far, so cost data are sparse. However, the probability is that, while Pyroceram materials will be more expensive than glass, they will be "much less costly than stainless steel," the company stated.



Missile radomes made from Pyroceram before (left) and after heat treatment that induces crystal growth and turns glassy noncrystalline material into opaque crystalline substance.

A few months after its discovery more than a year ago, Corning was awarded a Navy nose cone research and development contract by the Applied Physics Laboratory of The Johns Hopkins University. The missile radome is the result.

Lighter than aluminum and with a flexural strength of 40,000 psi, the Pyroceram used in the radome protects the sensitive directional instruments in the nose of the missile from the high temperatures of hypersonic flight. Corning was unable to say at this time which missiles will carry the radome.

High frequency insulating dielectric properties comparable to those of the best electrical ceramics have been found in one Pyroceram. At  $10^{10}$  cps the loss factor of this type is about two-thirds that of dense alumina.

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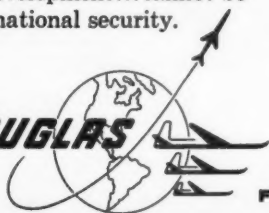
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# Jet Propulsion News

## Holaday Succeeds Murphree as Missile Czar

**W**ILLIAM M. HOLADAY, former Socony-Mobil Oil Co. executive, last month was named by Secretary of Defense Charles E. Wilson to succeed Eger V. Murphree as Special Assistant for Guided Missiles in a move designed to speed ICBM and IRBM development. Murphree will continue to serve the Defense Department in an occasional advisory capacity.



William M. Holaday

In announcing the appointment, Wilson also made known an administrative reorganization to hold down costs of the Vanguard earth satellite program, which have already soared to \$57 million, against an original outlay of \$20 million. Wilson stressed that the Vanguard project, as well as an unspecified program for development of missiles of moderate ranges, must not usurp priorities granted to long-range missile development.

Under the new setup, Holaday's responsibilities will include not only long-range ballistic missiles and the Vanguard project, but also any other ballistic missile program with ranges equal to or greater than the Army's 200-mile Redstone.

The directive to Holaday restates the highest priority for the five intercontinental and intermediate range missiles. It also gives him authority over the anti-ballistic-missile missile, as well as over guided missile ranges and their utilization.

The designation of Holaday to coordinate the earth satellite program

was said to be based on financial difficulties. However, officials stated that actual work on the project would not be affected, and launching is still planned for some time next year.

## MISSILES

- A record in speed—more than 9000 mph—is reported to have been reached by a Lockheed X17 at Patrick AFB. The three-stage research missile, powered by five Thiokol rocket engines, attained a speed of 9240 mph, or about 14 times the speed of sound. The previous record speed reached by a test missile was 8000 mph, recorded by a five-stage rocket fired from Wallops Island, Va. The X17 is also reported to have reached an altitude of 600 miles, the same as that reached by Jupiter C missile at Cape Canaveral, Fla. The records reported have not been officially confirmed.

- At the U. S. naval establishment, Roosevelt Roads, Puerto Rico, a center for missile firing training is being developed. First missiles, with a range of about 500 miles, will probably be fired within a year.

- The Air Force is expected to place limited production orders for the Bomarc anti-aircraft missile with Boeing Airplane Co. in the very near future. One contract, and possibly two, will shortly be announced by the Air Materiel Command, Dayton, Ohio, for the surface-to-air ramjet missile. While no official AF statement has been made about the size of the contracts, dollar volume for 1957 for the two contracts will probably exceed \$50 million.

- Western Development Division, Air Research and Development Command,

Inglewood, Calif., has been redesignated Headquarters, ARDC, Ballistic Missile Division. No changes in mission or organization are involved in the name change, made to provide a more descriptive title for the division in keeping with the expanded scope of Air Force responsibility in the ballistic missile field. Maj. Gen. Bernard A. Schriever will continue to command the newly designated headquarters.

- Staged by the Air Proving Ground Command of the USAF to provide comprehension of the combat-ready capabilities of air power, an aerial firepower demonstration was put on for 6500 visitors from all parts of the country and abroad at Eglin AFB, Valpariso, Fla., on May 6. Techniques demonstrated included air-to-air rockets, precision toss bombing by a B-57 Intruder, and jet aircraft launching rockets against simulated hostile gun batteries.

- Missile pads are under construction at Eglin AFB. Missiles will be tested in operational environment to develop techniques and demonstrate the ability of small crews to operate and maintain the equipment in simulated combat. Eglin, with its surrounding 465,000 acres, is headquarters for the Air Force employment and suitability testing agency. Last year construction was started of a launching site for ground-to-air missiles. The existing support facilities and instrumented land and water ranges make Eglin a natural choice for testing these missiles in operational environment.

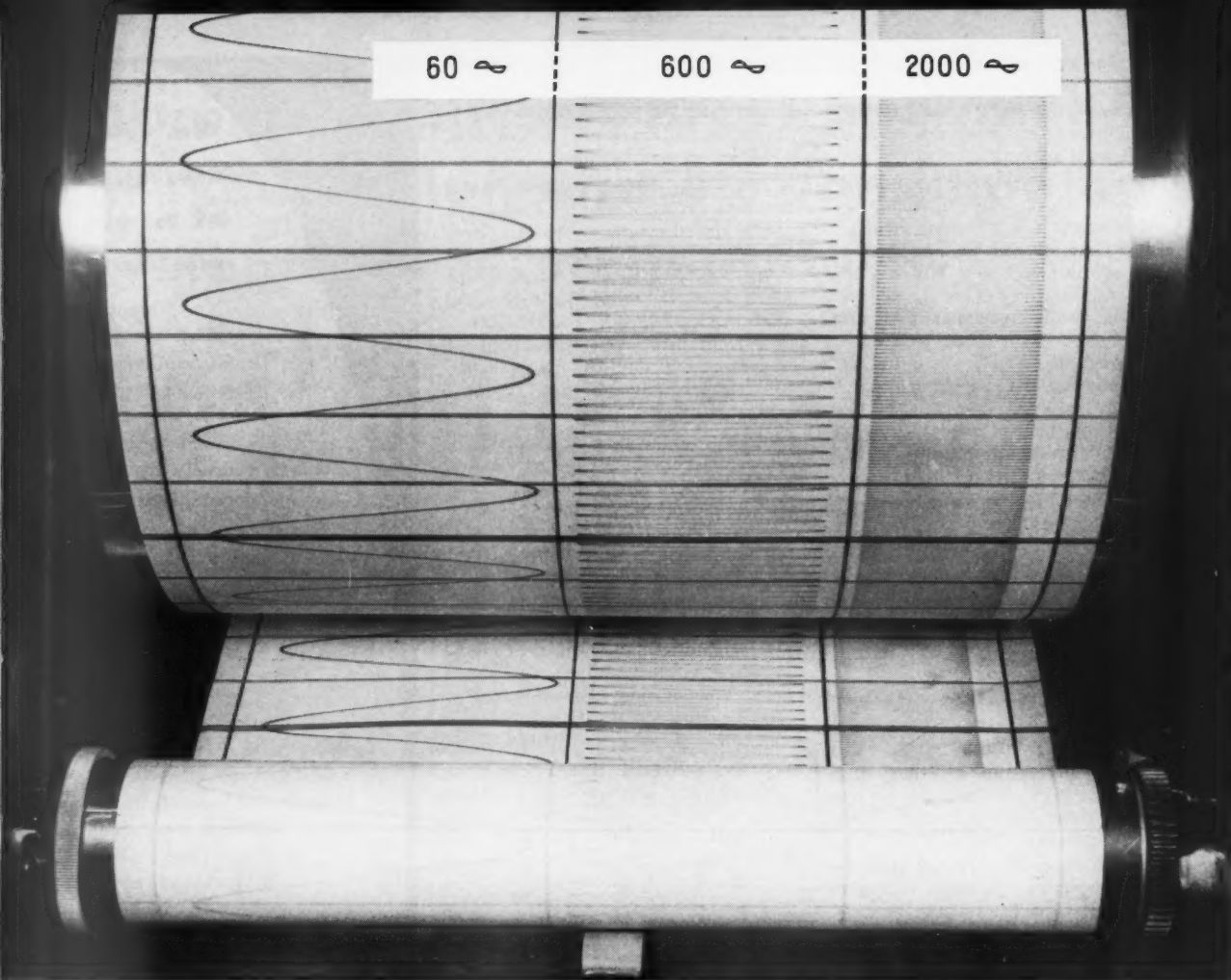
- The U. S. has proposed that all research on intercontinental ballistic missiles be stopped in the first stage of any



## New Air Force Missile Training Base

Ground-breaking ceremonies were held last month at Cooke Air Force Base, formerly Camp Cooke Military Reservation, Lompoc, Calif., which will be used

as a training base for Air Force missile units. Some AF personnel are now on the base, which is undergoing rehabilitation, modernization and construction.



Du Pont Lino-Writ 4 emerges from drying drum of an automatic processor after normal development. Unretouched photo shows clear, clean traces obtained from 60, 600 and 2000-cps traces recorded on *one* paper.

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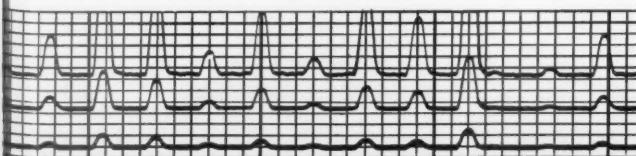
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disarmament agreement. In London disarmament talks, Harold E. Stassen suggested the big powers agree now not to build missiles and to open research to international inspection.

- To subject intermediate range ballistic missiles to the same motions they would encounter in launching at sea, a missile platform will be built for the Navy at the missile test center, Cape Canaveral, Fla. It will use gimbals and hydraulic pistons to simulate pitch and roll of a ship. Built by Baldwin-Lima-Hamilton Corp., it will be used for some land tests of the Lockheed Polaris IRBM.
- Initial launchings of the HASP weather rocket, fired from a 5-in. naval gun to 100,000 ft, were carried out in March. Further large-scale test launchings of the high altitude sounding projectile are being carried out by the Naval Ordnance Lab., Silver Spring, Md.
- To check methods of high altitude separation, tests were carried out of the launching vehicle for the U. S. earth satellite at Cape Canaveral. Only two stages of the three-stage rocket were involved in the test, the Navy announced. For the first stage, a Viking (14th and last to be made by The Martin Co.) was used. Second stage was a medium-sized solid propellant rocket similar to the one to be used for the third stage of the satellite launching vehicle.
- To correct any impression that the Cajun propulsion unit is an expensive rocket, H. Griffith Jones, general manager of Thiokol Chemical Corp., advises that his company offers the unit as an off-the-shelf item for \$1290 each, representing only 35 per cent of the cost of the Nike-Cajun rocket.
- IBM has delivered the 704 unit to be used by ONR to track and compute Vanguard satellite data.
- With improved Nike systems gradually replacing conventional antiaircraft artillery, the Army has changed the name of its major force engaged in defending U. S. cities. Formerly known as Army Antiaircraft Command (ARAA-COM), it is now the U. S. Army Air Defense Command (USARADCOM).
- At a meeting of the Arnold Air Society of the Air Force Reserve Officers Training Corps in New York, Thomas G. Lanphier Jr., vice-president of Convair, said he believes the U. S. and Great Britain are relying too much on early development of ballistic missiles. Referring to the British decision to rely principally on missiles for defense, Mr. Lanphier said that this policy apparently assumed that American-made missiles would be available in about a year. The missiles, however, may actually be several years away, he observed.
- At the Turbojet Engine Hydraulics Symposium held by Vickers, Inc., at

Detroit in April, Vickers exhibited pumps which have operated successfully at 450 F for 200 hours. The company also announced that pumps to operate at 550 F are six months away.

- Three new rocket-propelled target missiles were introduced by Radioplane at the guided missile symposium, Fort Bliss, Tex., in April. The fastest missile is the RP-76, designed to travel near Mach 1 at 50,000 ft; flight time, 10 min. Another missile, the RP-77D, a radio-controlled target drone powered by a Boeing 502-10F turboprop engine, has a speed of 400 mph.
- Nike-Hercules missiles with atomic warheads are defending many large U. S. cities, replacing the Nike-Ajax installations which use conventional warheads. One of the first of the new batteries, controlled by the "missile master" system, is located on Governors Island in New York.
- Planned for Formosa by the Air Force, a missile unit capable of reaching targets several hundred miles within the Chinese mainland will be equipped with Matadors. The tactical weapon is capable of carrying either a conventional or atomic warhead.
- Army Secretary Brucker told an audience in Chicago that nuclear weapons have become cheaper than conventional explosives. This fact has brought about a major shift to atomic arms, particularly guided missiles, in the U. S. and Russia.
- Navy contracts totaling \$51.5 million were received by Sperry Rand for a radar to guide Terrier missiles. Two contracts totaling \$23 million went to Northern Ordnance, Inc., to produce launching equipment for the Terrier. These systems will be used on two new supercarriers and a new class of guided missile frigates, and for launching systems for the first atom-powered guided missile cruiser authorized by Congress.
- GE has a \$1.5-million contract for developing the fire control system to explode the nuclear warhead of the Navy's Polaris IRBM, the 1500-mile range missile now under development.
- North American's missile development division has received an additional \$21 million for research and development of the Air Force SM-64 Navaho intercontinental strategic weapons system. Testing of the Navaho began after completion of the flight program of the X-10 interim test missile.
- The world's largest solar furnace will be built by USAF's Air Research and Development Command near Cloudcroft, N.M. Producing temperatures up to 8000 F, it will be used to study the effects of rapid temperature changes on missile structure metals.
- Between late 1954 and this July, \$300 million will have been spent in new in-

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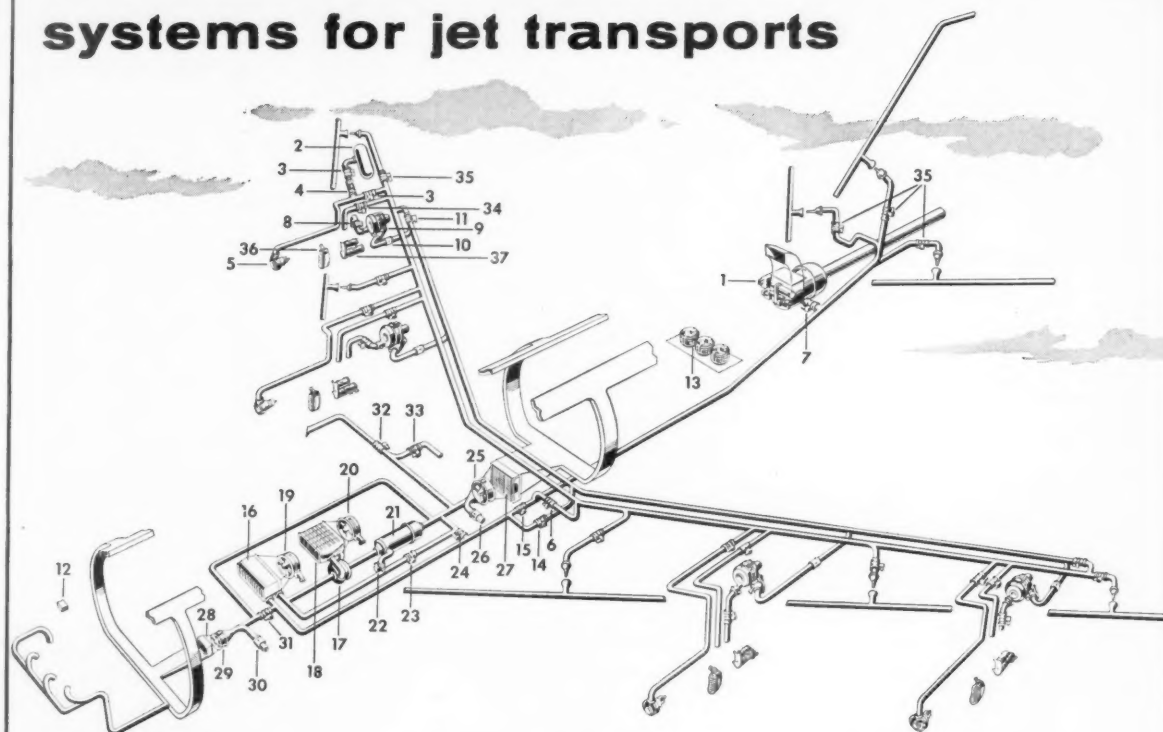
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dustrial facilities for the ballistic missile program. Of this sum, one-third was provided by the contractors.

- AEC announced that the U. S. would fire its first air-to-air atomic rocket during the nuclear tests held last month in Nevada. The rocket, armed with a nuclear warhead, was to be fired from a piloted plane.

## AIRCRAFT

- High speed aircraft are launched from short fields by a unique air-transportable jet-powered catapult currently operated by the U. S. Marine Corps. In a demonstration before Navy observers, the turbo-catapult launched jet aircraft in less than one-fifth their usual take-off run. The "turbo-cat," developed by All American Engineering Co., Wilmington, Del., develops 60,000 hp.

In the recently patented device (see *JET PROPULSION*, Jan. 1957, p. 82), six jet engines are arranged in a circle with the exhaust gases flowing into the central launching turbines. The turbines operate a drum cable system driving a shuttle guided in a track; the aircraft is attached to the shuttle by means of a conventional bridle.

- Three F-100C Super Sabre jets last month flew 6710 miles nonstop from London to Los Angeles in the longest single-engine jet flight ever made. The 14-hour, 5-min trip was described as a "routine, long-range, cruise control flight."

- Trans-Canada Air Lines has ordered two additional Douglas DC-8 jet transports, exercising its option to increase its original purchase of four long-range 550-mph jetliners. Delivery of TCA's jet fleet will begin in 1959, and service over trans-Atlantic and transcontinental routes will start in 1960.

The DC-8's for the Canadian airline will be powered by Rolls-Royce Conway turbojet engines of the by-pass type. They will carry 120 passengers from Montreal to London in 6 hours and 10 min.



- A plastic model of a new "aerial bobsled" supersonic ejection seat (photo) is being tested in a Southern California cooperative wind tunnel. The model is projected by compressed air into the

tunnel windstream. The "A" seat is one of two basic approaches being tested and evaluated for "century series" pilot escape systems by the Industry Crew Escape Systems Committee, an industry-wide cooperative program to develop for the Air Force a standardized safe ejection system for jet aircraft.

- Wind tunnel tests are nearly completed and fabrication has started on the X-15 research airplane at North American's Los Angeles plant. Reaction Motors is making the liquid propellant rocket engine.



- One of the world's smallest helicopters, the Ultra-Light jet, has been introduced in the U. S. by Piasecki Aircraft Corp. representing the Fairey Aviation Co., Ltd., of England (photo). Built to the British Army's front-line duty specifications, the two-seat, jet-powered, single-rotor helicopter is under evaluation by the U. S. Defense Department. Piasecki has an option to manufacture the helicopter in the U. S. It will be known here as the company's Model 72.

- The Ryan X-13 Vertijet, the U. S. Air Force's jet-powered vertical take-off and landing research airplane, accomplished its first complete operational flight sequences from vertical take-off, transition to horizontal flight and return to vertical landing. Two flights were made at Edwards AFB where the demonstrations culminated an 18-month flight test program.

- First landing of a commercial jet airliner at N. Y. International Airport was made by the French Caravelle after a flight from Europe by way of Africa and Brazil and then to Miami. The twin-jet airliner, designed and built by Sud Aviation, visited in N. Y., opening a demonstration tour of 16 cities of the U. S. and Canada. Production models will have 11,500-lb thrust Rolls-Royce R.A. 29 Avons, and will seat 64 first-class or 80 tourist-class passengers.

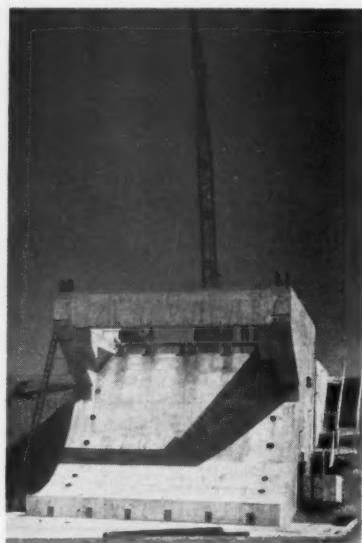
- Upon his return from a sales tour of Western Europe, Wellwood E. Beall, Boeing senior vice-president, predicted European airlines may double the number of jet transports they now have on order. European lines have contracted for 40 Boeing 707's and 22 Douglas DC-8's. Deliveries of 707's will begin late in 1959. Boeing has ten additional overseas orders—three for India and seven for Australia.

- TWA became America's first commercial jet airplane operator when its jet and propeller powered Fairchild C-82 was commissioned at Orly Field, Paris. The "flying maintenance base" C-82 carries a 1000-lb-thrust Fairchild J-44 turbojet engine installed on top of the fuselage for jet-assist take-off. The aircraft is used solely as an engine carrier and mobile base to aid in maintaining on-time schedules along its foreign routes.

- Flight tests of a new "pure" inertial navigating system for automatic guidance of manned or unmanned aircraft were announced by Minneapolis-Honeywell Regulator Co. The system, designated ISIP (inertial system indicating position), does not depend on outside sources such as human pilot, radar, star tracking or radio commands. Since it does not accept nor emit signals from outside sources, it cannot be detected by known devices, nor can it be jammed by an enemy.

## COMPANIES

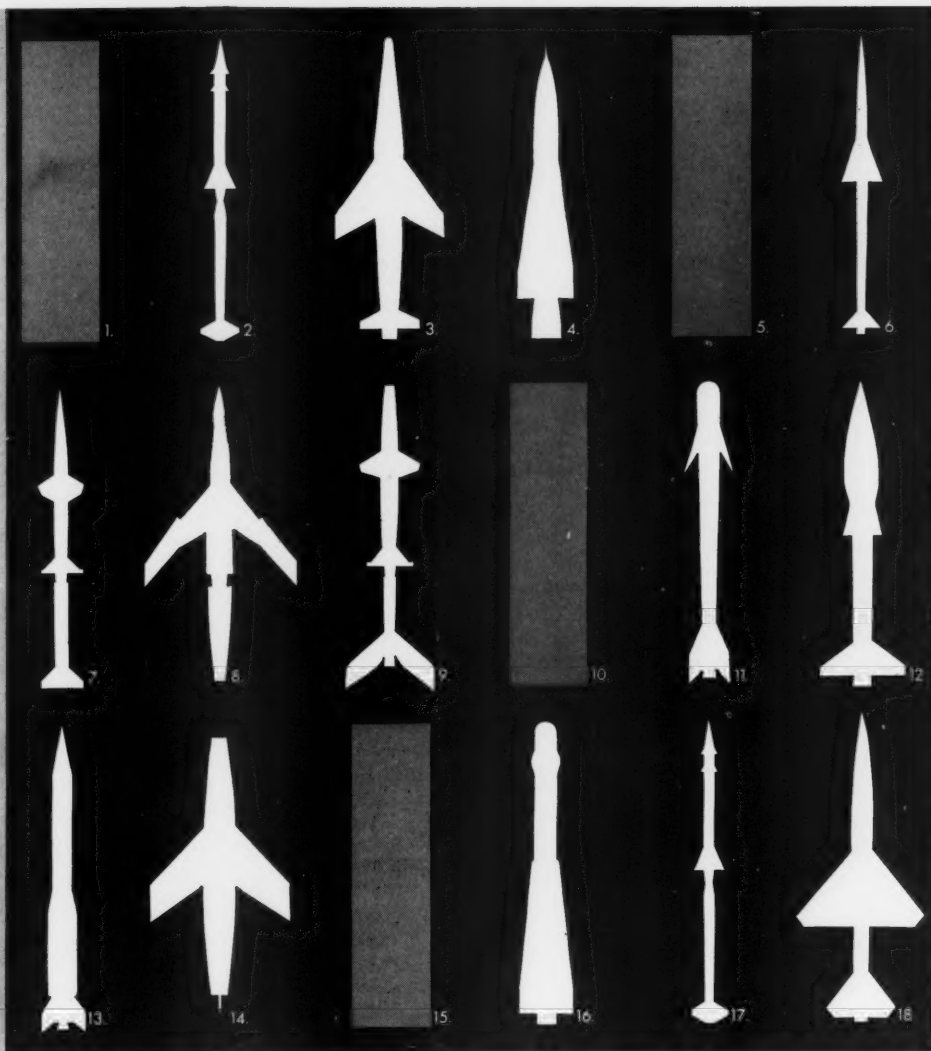
- Rocket engine production has started at the new \$13.2-million Rocketdyne plant at Neosho, Mo., following the award of an Air Force contract for an undisclosed number of propulsion sys-



tems for Thor IRBM. Built in support of the AF ballistic missile program, the Neosho plant, occupied last December, includes a 200-acre test firing complex (photo shows one of the test stands before installation of steel superstructure) and 228,000 sq ft of manufacturing and office space.

- A \$4,595,600 contract was awarded to Pittsburgh-Des Moines Steel Co. by the USAF for construction of three removable wind tunnel test sections at Arnold Engineering Development Center, Tullahoma, Tenn. Each unit is 20

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ft long and self-propelled over a railway.

- In keeping with the industry's recognition of the impact of foreshortening company names, The Glenn L. Martin Co. has changed its name to The Martin Co.
- Around - the - clock production of guided missile and jet hardware began at the new Franklin Park, Ill., plant of Diversey Engineering Co.
- California operations of the Aero Hydraulics Div., Vickers, Inc., have moved from El Segundo to a new plant at Torrance to accommodate engineering, sales service and production of aircraft, missile and engine hydraulics systems and components. Administrative headquarters for the division continue in Detroit.
- The cornerstone was laid for a million-dollar plant for Servomechanisms, Inc., in Westbury, L. I. The company operates another plant in Westbury, and also one in Hawthorne, Calif. A wholly-owned subsidiary, Servomechanisms (Canada) maintains facilities in Toronto.
- Era Engineering, Inc., Santa Monica, Calif., has been formed by a group of scientists and engineers to develop new systems and devices in electronics, radiation and rocketry. Systems now available include a radioactive technique to mark and insure recovery of missiles after experimental flight tests.
- Task Corp., Anaheim, Calif., has an Air Force contract for a positive displacement pump to provide oxygen to pilots of supersonic aircraft.
- Burroughs Corp. plant in Detroit has been doubled in size by construction of a second manufacturing building. The company produces electronic data processing and transmitting equipment, and is a prime contractor in the USAF ballistic missile program, with contracts for processing equipment used in ground control guidance systems.
- Air Materiel Command awarded Kollsman Instrument Corp. a contract in excess of one million dollars for pilot production of the AN-AVN-1 Astro Navigational Set known as the Astro Tracker.
- U. S. Industries, Inc., New York, has acquired the outstanding stock of Kett Corp. of Cincinnati, a research and development firm in the aircraft and missiles field. The company will spend \$6 million this year to complete its capital expansion and improvement program.
- An order has been placed with Southwestern Industries, Inc., Los Angeles, for the design and manufacture of pressure switches for missile applications, by Convair-Astronautics in San Diego.
- Trans Continental Industries, Inc., Detroit, has established a division to make hardware for guided missiles.
- U. S. Flare Corp. and Associates, Pacoima, Calif., producer of rocket ignition systems and missile tracking systems, has been acquired by Atlantic Research Corp., and will operate under its own name as an Atlantic affiliate.
- A 22-million-volt Allis-Chalmers Beta-tron will be installed in Electric Steel Foundry Company's Portland plant, complementing the present 220,000-volt x-ray and Cobalt 60 radiographic facilities now in use.
- Universal Atomics Corp. has become Universal Transistor Products Corp., with plant and executive offices at 50 Bond St., Westbury, L. I., N. Y.
- American Electronics, Inc., has established a division called American Laboratories to produce facilities for testing electrical, electro-mechanical and electronic packages up to 10 cu ft, at Fullerton, Calif. Test areas will be air conditioned, humidity controlled and pressurized, and will have a 4-ton liquid CO<sub>2</sub> distribution system.
- Minneapolis-Honeywell acquired Rubicon Co., Philadelphia, makers of precision potentiometers, bridges and other devices for missile instrumentation.
- Farnsworth Electronics Co., Fort Wayne, Ind., a division of I. T. & T. Corp., has received from Boeing Airplane Co. additional orders amounting to more than \$12 million.
- Missile-Air, manufacturer of flush latches and precision aircraft and missile components, has moved its engineering and production facilities to new quarters at Gardena, Calif.
- The Rheem Aircraft Div. has been formed within the Rheem Mfg. Co. to handle diversified programs in the aircraft, jet engine and ordnance fields. The division replaces the former Government Products Div., maintaining headquarters in Downey, Calif.
- Union Carbide and Carbon Corp. has changed its name to Union Carbide Corp. Divisional name changes include Carbide and Carbon Chemicals Co. to Union Carbide Chemicals Co., and Linde Air Products Co. to Linde Co.
- A 30-year-old name receded into the background when Air Associates, Inc., officially became Electronic Communications, Inc. The company is undergoing gradual relocation which will place all manufacturing and engineering operations in St. Petersburg, Fla., by fall.
- BuOrd awarded a \$3.5-million contract to Aerojet-General for perchlorate oxidizer JATO units and spare igniters.
- Guided missile production facilities have been added to the Missile and Ordnance Systems Dept. of General Electric, supplementing its research and development capabilities. The department now occupies three major locations: Research and development in Philadelphia, ordnance in Pittsfield,



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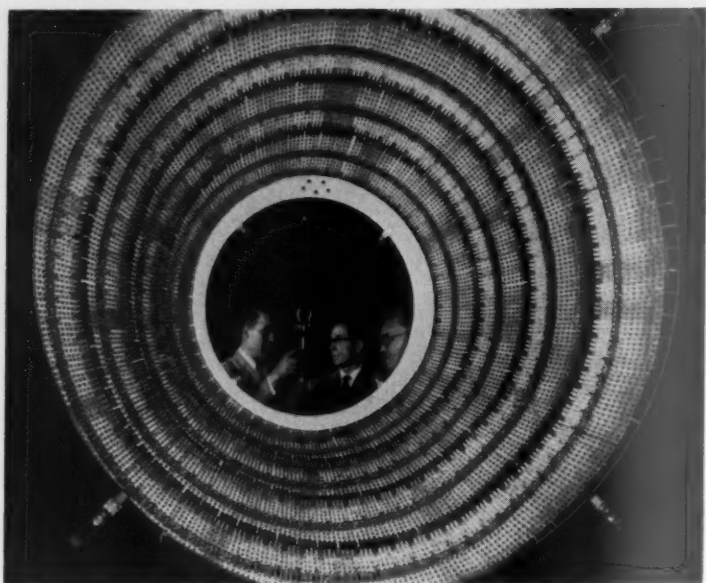
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### ... And Up Pops Redstone

This giant missile "toaster," consisting of more than 1000 quartz infrared lamps arranged in closed banks to form a cone shape, has been designed by Chrysler for simulating re-entry temperatures encountered by the Army's

Redstone surface-to-surface missile. Thousands of degrees F with almost no warm-up period are claimed for the unit. One test consumes 3,600,000 watts (enough to toast 7000 slices of bread, the company figures).

and missile production in Burlington, Vt.

- Friez Instrument Div. of Bendix Aviation Corp. is expanding its facility in a million dollar program at Baltimore, Md. Among products to be made are electromechanical missile components.
- Newly formed Physical Measurements Corp., Santa Monica, Calif., will produce accelerometers, switches and transducers.

### INSTITUTIONS

- Second in a series of bi-annual symposia in gas dynamics is scheduled for Aug. 26-28 at The Technological Institute of Northwestern Univ. Its theme will be "Transport Properties in Gases at High Temperatures and Pressures." Address inquiries to Dr. Ali Bulent Cambel, Gas Dynamics Lab., Northwestern Univ., Evanston, Ill.
- The National Science Foundation has announced 235 grants totaling \$4,316,352 awarded during the first quarter of 1957. Grants are for the support of basic research in science, exchange of scientific information and training of science teachers.
- The 14th annual display of aviation electrical equipment conducted by the Aircraft Electrical Society will be held Oct. 24-25 in Pan Pacific Auditorium,

Los Angeles. It is estimated that 200 equipment manufacturers will participate.

- T. Paul Torda of Polytechnic Institute of Brooklyn is working on a project for investigating combustion instability and "scaling-up" of rocket motors, under contract with the Air Force Office of Scientific Research.
- More than 2000 members and their families are expected for the 65th annual meeting of the American Society for Engineering Education, June 17-21, Cornell University. Following the meeting, the university will be host to a ten-day workshop for staff members of the technical institutes and community colleges.
- A new group to deal with the increasingly complex engineering problems encountered in keeping industrial equipment in operating condition has been formed by the ASME. The unit will cover such topics as modification of equipment to reduce maintenance, and managerial activities.
- The El Paso *Herald-Post* and El Paso *Times* will publish special feature editions on Saturday, Aug. 31, and Sunday, Sept. 1, on the role of the Southwest as a research and development center of the astronautical, as well as atomic and nuclear, sciences. The editions will detail the roles of White Sands Proving

Ground, Holloman Air Development Center and Fort Bliss in the over-all rocket and guided missile picture.

### FOREIGN

**Australia:** Defense Minister Sir Phillip McBride said Australian policy is based on the integration of land and sea weapons with those of America. In this connection, his government has decided to buy 30 Lockheed F 104 Starfighters and 12 Lockheed C-130A Hercules transports.

**Canada:** Canadair Ltd., Montreal subsidiary of General Dynamics Corp., announces that a new version of the Britannia will soon be available. Re-entering the commercial aviation field after an absence of seven years, Canadair offers its own version of the Britannia under an agreement with Bristol Aeroplane Co. of England.

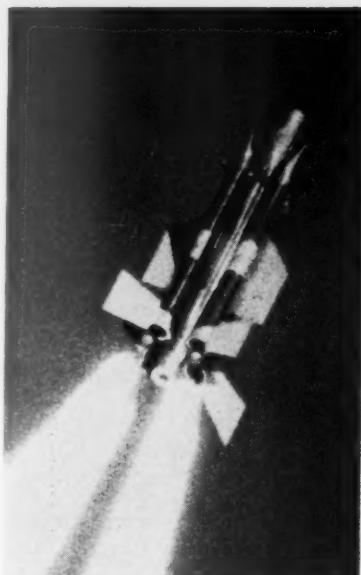
Known as the CL-44, the plane will be powered by four Bristol Orion turbo-prop engines, and will have airliner, freighter and military transport configurations.

**England:** The Engineer Research and Development Lab. of the U. S. Army placed a contract with D. Napier and Son Ltd. for a diesel engine compressor set for use with a mobile liquid oxygen plant. It will supply up to 30,000 lb of dry air per hour at 100 psi.

- Brief data have been released on the Bristol Bobbin missile. This vehicle has been used for the development of the Thor ramjet engine, designed as a powerplant for the Bloodhound missile. Twin 16-in. Thor units are mounted on each side of the missile, which uses four booster rockets. Thrust of the Thor is said to be 15,000 lb at Mach 3 at sea level. The Bobbin is probably the first British-designed recoverable test vehicle.

- The Minister of Supply recently announced that production orders had been placed for four guided weapons. Until now, the only production order announced had been for the Fairey Fireflash air-to-air rocket, an RAF training weapon.

The three new weapons are the Bristol Bloodhound, a ground-to-air rocket-boosted ramjet; the de Havilland air-to-air rocket Firestreak; and the as yet unnamed English Electric ground-to-air rocket (photo). All that is known of Firestreak is that it has an infrared homing device capable of homing on targets up to eight miles or more away. No performance details were given of the Bloodhound except that its rocket boosts it to 1000 mph. More than 200 of the vehicles have been fired at Averborth, on the Welsh coast, and on the Woormera Range in Central Australia.



• The Sea Slug missile developed for the Navy by Armstrong Whitworth was not mentioned by the Minister of Supply, probably because this weapon requires a fully stabilized launching platform and control. Further trials are likely first, both at sea and on the land-locked Clausen Rolling Platform.

• Another missile mentioned was the "stand-off" powered bomb that Avro is developing for use with the V-class heavy jet bombers Vulcan and Victor. It was said to be about the size of a small fighter.

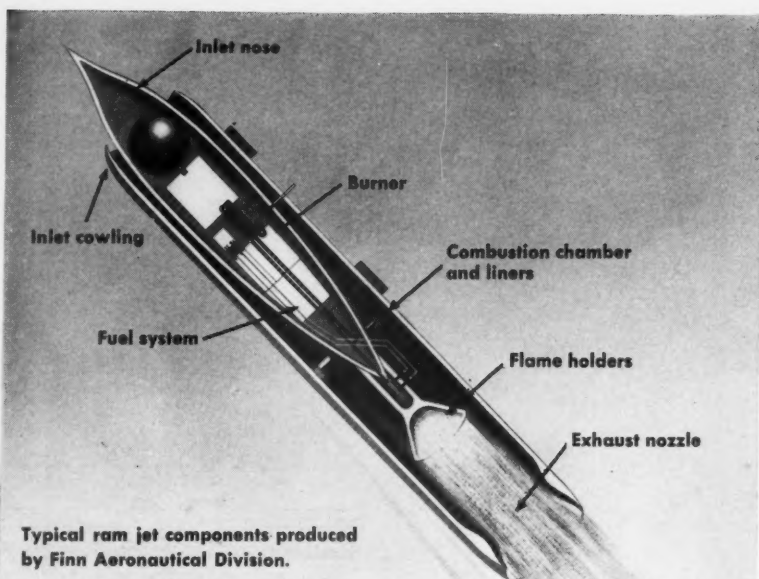
**France:** More than 30 flights have been made with the Leduc 022 using its Atar turbojet engine only. The ramjet engine will also be put in operation during the next test phase.

• A Sud-Aviation Djinn helicopter, powered by a Turbomeca Palouste turbine and flown by Jean Dabos, reached a world's record altitude of 27,829 ft.

**India:** The Indian Air Force received initial deliveries of 125 Dassault Mystere IVA jet fighters. The French aircraft were purchased despite the fact that the Russians offered MiG-17's at a most attractive price.

**Israel:** It is reported that the French guided missile S-S 10 was tested in battle by the Israel Army against soviet-built tanks last fall in the Sinai Desert campaign. French Air Force Gen. Daun said that the 12-lb charge in the missile can destroy any tank, and has a record of 90 per cent hits.

**Japan:** Three solid propellant missiles are being tested at the Soma-garah range, Central Japan. They are about 6 ft long and have swept-back wings. The first firing was unsuc-



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cessful and the missile crashed after ascending to 150 ft.

### RESEARCH & DEVELOPMENT

- Crucible Steel Co. of America renewed its contract with the USAF Materials Lab., Wright Air Development Center, for development of new high-temperature steels for jet aircraft.
- Ground was broken by Sylvania Electric Products, Inc., for a multimillion-dollar research and development center at Amherst, N. Y., for the company's Electronic Systems Div. Construction was also started on a plant for Waltham Laboratories, part of the Electronic Systems Div., at Waltham, Mass. The plant will house the Missile Systems Lab., now occupying part of a building completed two years ago.
- A new temperature-altitude-humidity walk-in test chamber has been completed by American Research Corp., Farmington, Conn., for simulated flight testing of electronics components. Temperature range is -100 to 300 F; relative humidity, 20 to 100 per cent over a dry bulb range of 35 to 185 F; altitude, up to 100,000 ft.

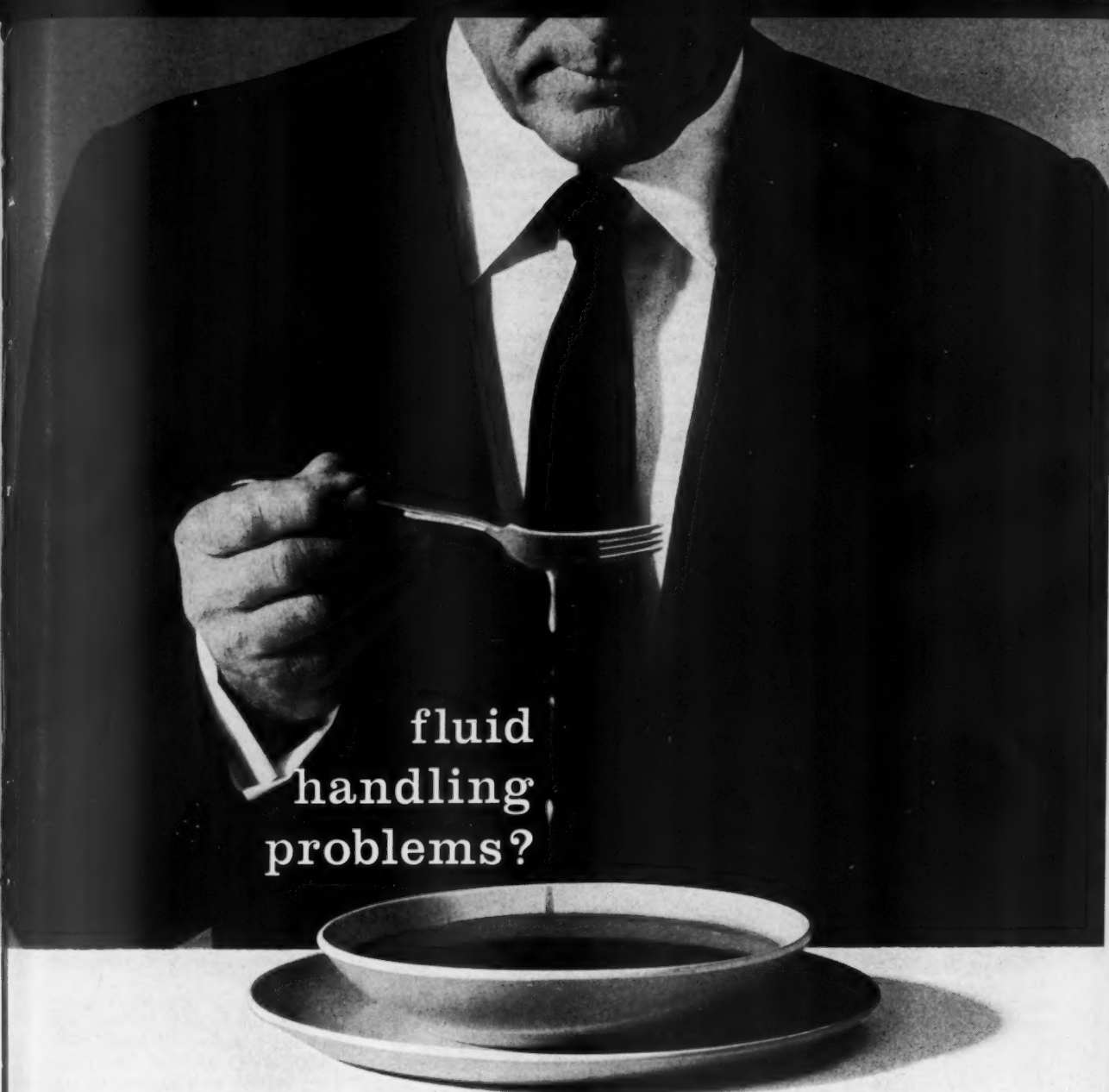
### Space Medicine Problems Spotlighted

A session on space medicine and a space travel symposium bringing together some of the nation's leading figures in the field of astronautics highlighted the 28th annual meeting of the Aero Medical Assn. at Denver, Col., last month.

Participants in the space travel symposium, under the chairmanship of Col. Paul A. Campbell, USAF (MC), examined almost every aspect of extra-terrestrial flight.


Wernher von Braun, ABMA, Huntsville, Ala., addressed the group on the propulsion engineer's view of manned extra-atmospheric flight, while Walter O. Roberts, High Altitude Observatory, University of Colorado, Denver, presented the astronomer's views. The astrophysicist's and test pilot's views were presented, respectively, by Heinz Haber, University of California at Los Angeles, and Scott Crossfield, North American Aviation, Los Angeles.

Comdr. George W. Hoover, Office of Naval Research, Washington, D. C., discussed the problems involved in instrumenting space vehicles, while John P. Hagen, NRL, talked on the space travel implications of the Vanguard project. The survival aspects were discussed by Alfred M. Mayo, Douglas Aircraft Co., El Segundo, Calif., and Dr. Hubertus Strughold, School of Aviation Medicine, Randolph AFB, Tex., chose as his topic the possibilities of reaching an inhabitable extra-terrestrial environment from earth.



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## Chemical Men Probe Missile Business

**L**AST month, the Commerical Chemical Development Assn. held its Spring meeting in French Lick, Ind. Purpose of the symposium, as stated in its topic, was to find out "What the Rocket and Missile Program Means to the Chemical Industry."

For those chemical men who had little idea what missiles were, the Indiana meeting provided a big, booming answer: A \$2-billion-a-year business that is going to get even bigger. For the missile industry, the meeting had a different significance.

**Still Waiting:** In essence, this was a meeting of the chemical industry's top dollars-and-cents men devoted solely to exploring the market potential of the missile field. In effect, it could mean that the chemical industry is collectively taking a more active interest in the missile field and, perhaps, will initiate, speed up or expand its long-awaited technical collaboration with the missile industry. Heretofore, the chemical industry's effort in behalf of the missile program has been at best a spotty one, with comparatively few companies actively seeking an important role in the field.

Everyone agreed that in the light of the chemical industry's position as a material processor, supplier and user, and as a high temperature process operator, it can contribute a great deal to the missile program. But for the market men at French Lick the big question was: Is it worth it?

**Market Trajectory:** Any new chemical venture, of course, requires a sizable investment of time and money. And, before making any commitment, a company generally tries to insure a pay-off. This is the job of the market men.

From facts, figures and informed speculation, they try to develop a picture of the potential market. In rough focus, the picture for a "worthwhile" venture should show a yearly demand of at least 10 million lb (a lower figure is acceptable if a large enough potential is evident) and a life span long enough to return the initial investment, operating expenses and a fair profit.

At the CCDA meeting, market men were told that the facts and figures were available, provided they were willing to go to the trouble of getting them. Obtaining security clearance was recommended as the first step. At the meeting itself, they learned and often were confused by the extent of the informed speculation present in this area. And potential market dura-

tion, they were told, would prove almost impossible to estimate because no one, even in the missile business, had any definite ideas as to what could be expected.

In dollars, market estimates were pegged as high as one-third of the over-all missile budget, or approximately \$65 million/year. These covered all possible missile expenditures falling within the purview of the chemical process industries; e.g., research, development, nonferrous metals, plastics, ceramics, propellants, hydraulic fluids, lubricants, pressurants and control elements such as germanium.

In a more conservative vein, Thiokol's H. W. Ritchey estimated that the propellant share of the total missile budget ran from  $\frac{1}{2}$  to 5 per cent.

Pennsalt's John Gall figured that peak peacetime demand for a propellant "may soar to hundreds of tons per month," but called it an awkward market. Gall likened the demand curve for missile chemicals to the trajectory of a missile itself. It starts with a few pounds in the preliminary phase, moves to a few dozen pounds in the large laboratory phase or, if not too expensive, to a few hundred pounds; hits a peak in the hundreds of tons per month during the testing program; and then drops off to a small amount for replacement, training and minor development programs.

**A Starting Point:** As unattractive as this demand trajectory first appeared, some market men soon became convinced that it merited further study. They reasoned that, if a firm had any desire to get into a new chemical field, the chemical's missile market would probably defray a sizable portion of the large initial investment required for starting any new venture.

Another possibility, as pointed out by Dr. Gall and other speakers, is for a company to enter a new chemical venture solely on the strength of the missile market, and then try to develop commercial outlets during the years of the missile market. Admittedly a risky course, some seemed to believe it would be worth the gamble more often than not.

At this point, no one was willing to guess just how many additional chemical companies would be attracted to the missile field by these market possibilities. The general feeling, however, was that the meeting had been successful in giving most of the 200 conferees a clearer picture

of the missile program and their relationships to it.

**Interesting Reflections:** Designed primarily as an unclassified introduction to the market potential for chemical propellants in the missile program, the meeting had little new information to offer the enlightened. Most missile men were there to teach, not to learn. But since the teachers came from some of the leading missile firms and the more missile-minded chemical companies, the meeting—particularly in the follow-up discussion periods—offered a good reflection of top-level thinking now current in the field.

On the use of a liquid fluorine in rocket propellants, for example, John Tormey of Rocketdyne calculated that a combination of fluorine and oxygen can provide a 50 per cent increase in specific impulse over present propellant combinations. Dr. Gall declared that fluorine is now ready for use in the rocket field, further asserted that its use in rockets is inevitable. (For more on fluorine, see p. 678).

Olin Mathieson's Earl Weilmuenster disclosed that, of all the boron-based high energy fuels, his company is most interested in pentaborane and decaborane from the standpoint of use in liquid propellants. He also revealed that along with the boron fuels, other new Olin Mathieson propellants are being evaluated for use in various new missile and aircraft systems. While Weilmuenster indicated that O-M wasn't interested in beryllium because of toxicity and supply problems, it was pointed out that "what has been done with boron can be done with beryllium" as far as processing goes.

To a question on materials, Tormey replied that the use of new propellants isn't being held back by any lack of adequate structural materials. Rather, he said, it is the mechanical engineer that is being delayed in the development of lighter weight engine components.

**Vis-A-Vis:** One of the most interesting discussions took place just before the close of the meeting when moderator Arthur Stosick of Union Carbide asked the speakers if they would care to comment on the merits of solid vs. liquid propellants. Here, briefly excerpted, is what followed:

TORMEY was willing to concede that solid propellants were preferable for small missiles but maintained liquid propellants were better for the very large vehicles. In between, he declared, is where the argument takes place.

RITCHEY disagreed, seeing no size limit for solid propellant rockets in military applications. True, he

said, if the rocket is to be recovered, then liquids would be better as a matter of cost. But a solid rocket can be made 20 times bigger than a liquid rocket, according to Ritchey, and still get off the ground without running into the problem of trying to pump a tremendous volume of liquid. (Earlier, Ritchey had said that with a fourth stage, Lockheed's solid propellant X-17 could serve as a satellite launching vehicle; and with a fifth stage, it could carry a small 2-4 lb device to the moon.)

RICHARD CANRIGHT of Douglas Aircraft contended that pumping would be no problem in a large liquid rocket, that an engineer simply could use additional pumps. In his estimation, solids were more reliable but less safe (in terms of personal danger) than liquids.

STEWART JOHNSTON, consultant to Ramo-Wooldridge, pointed out that solids presented more of a problem in thrust termination.

ROBERTSON YOUNGQUIST from RMI added that liquid rockets offer greater flexibility in controlling vector direction and thrust. And, he said, liquids have a greater inherent specific impulse potential.

ALLEN DESCHERE of Rohm & Haas conceded that what Johnston and Youngquist said was true now. But, he went on, solids are comparatively inexperienced in this area. In the long run, he was confident, solids would compete with liquids in controllability.

DAVID ALTMAN, Aeronutronic Systems moderator of the first session and a "neutral," pointed out that solid rockets were 20 to 50 per cent more compact than liquid rockets and therefore more rugged; that development costs of solids are greater but this is offset by lower operational costs; that for longer ranges and missions, liquids are more desirable because the liquid rocket can be better staged to get greater specific impulses.

In a final aside, Canright recalled a recent bull-session at Douglas in which missile engineers noted that both liquid and solid rockets have rather complex hydraulic and guidance systems which military personnel have to be trained to handle. Therefore, they felt the government could save money by employing a roving band of civilian technicians who would travel from one installation to another making sure the missiles were always in good condition.

The meeting ended on this note, with neither side convinced that the other had produced strong arguments but both definitely agreed that missiles are here to stay.

## New Sperry Radar Piped Aboard

A new class of high performance radar already in use aboard missile ships to guide Terrier missiles was revealed last month. Developed by Sperry Gyroscope and designated AN/SPQ-5, the radar is reported by the Navy to give "exceptionally high performance for tenacious, stable guidance of supersonic missiles, whether fired singly or in salvos . . ."

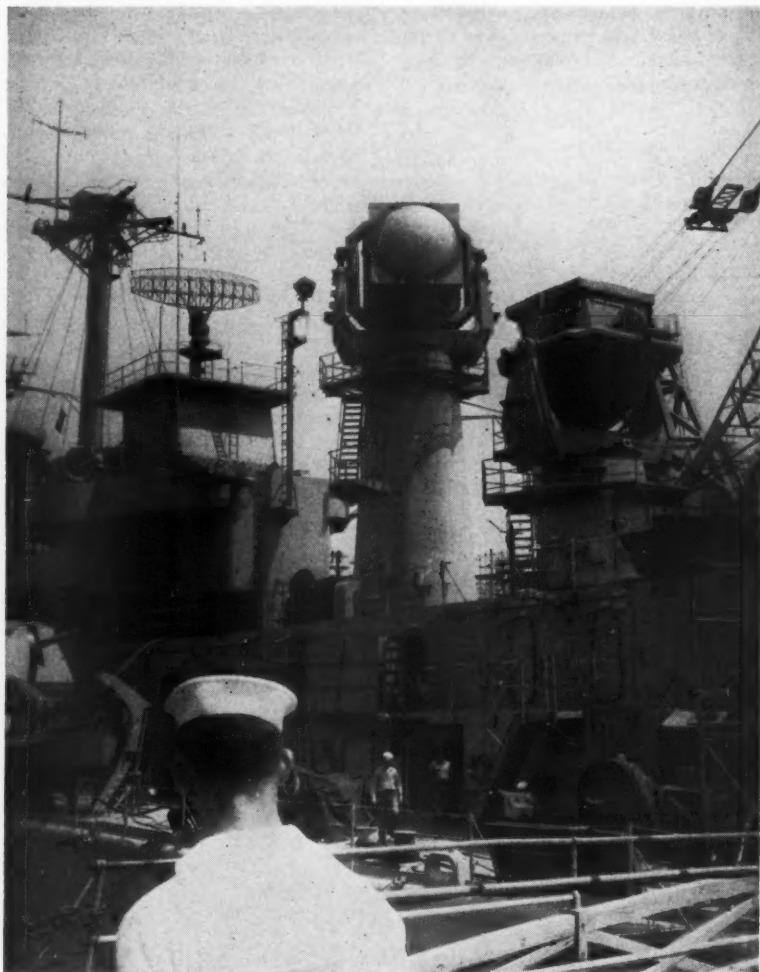
The long-range, high altitude missile guidance radar systems are a product of a Navy fleet defense program directed against supersonic jet aircraft, according to Rear Adm. F. S. Withington, chief of the Bureau of Ordnance. Years of successful testing preceded ship installation.

Two such radars are aboard the USS Canberra. Their antennas resemble giant searchlights. Additional features cited for the new system were

flexible modes of scanning the air space beyond the horizon, and the property of selecting and tracking individual targets in a close-flying group of supersonic attackers at great distances, while the missiles are launched and guided with "extreme accuracy."

At the same time, Sperry announced that a new manufacturing facility at Charlottesville, Va., the Sperry Piedmont Co., is producing the super radars for the Navy. Completed in October, the new \$2-million plant includes special facilities to accommodate the massive radar antenna "barbettes" used aboard ship.

The USS Boston, in addition to the Canberra, is equipped with Terriers and, presumably, the new radar systems. Three other cruisers are currently being converted, and eight frigates are being built to take Terrier launchers.



Giant searchlight-like antennas shown on the Terrier-equipped Canberra signal the use of a new Sperry radar system by Navy missile ships.

# Space Flight Notes

JOHN GUSTAVSON, Convair-Astronautics, Contributor

## The Red Planet

EVER since the days of Jules Verne, discussions of the possibility of interplanetary flight have centered on two close neighbors of earth—Mars and Venus. Of the two, Mars has undoubtedly received the most attention from scientists as well as science-fiction writers, possibly because of its unusual color, which makes it clearly distinguishable from other celestial bodies.

With space flight no longer in the distant future and Mars a likely destination for the first attempts at interplanetary travel, some consideration of the physical characteristics of the "Red Planet" may help give an idea as to the type of world which will greet the first space travelers.

<sup>1</sup>Ley, Willy, and von Braun, Wernher, "The Exploration of Mars," The Viking Press, New York, 1956, p. 27.

## Astronomical Facts

As Table 1 shows,<sup>1</sup> Mars is approximately one and one-half times farther from the sun than the earth. The apparent effect of this greater distance is a comparatively lower temperature on Mars. The solar constant at Mars' distance is approximately 40 per cent that of the earth. The high eccentricity of the orbit creates more pronounced seasonal conditions. The northern hemisphere experiences long summers and short winters; the southern undergoes correspondingly short summers and long winters. Consequently, the southern hemisphere shows a larger polar cap. The longer year also contributes to the exaggerated seasons.

It was hoped that the latest opposition, Sept. 8, 1956, could yield sufficient information to solve many surface feature problems. Unfortunately, oppositions are always at perihelion, i.e., when Mars is closest to the sun. It is at that particular time of the Martian

year that dust storms are churned up by the intense heat, and thus surface details at the last opposition were mostly obscured. However, other pertinent information was gained. The crux of this material, in the light of earlier observations, will be presented here.

## The Atmosphere

The composition of the Martian atmosphere has not been accurately determined. The dynamical theory of gases can exclude some components such as hydrogen and helium. These light gases disappeared long ago from the atmosphere as a result of the high mean molecular velocity and low escape velocity of the planet. Very reactive gases, such as ozone and chlorine, are excluded, as they will combine readily with surface minerals. The Martian atmosphere is at present believed to consist of 98 per cent nitrogen and some 2 per cent carbon dioxide, with some traces of oxygen and water vapor.

Pressure at the surface is comparable to the pressure at 11 miles altitude on earth, or more specifically 2.6 in. of mercury. The pressure gradient of the Martian atmosphere is much less than that of our atmosphere due to the lower gravity. Therefore, the Martian atmosphere must extend to greater altitudes. In light of this, the entry problem of a descending vehicle would be quite different from the re-entry problems already known to exist for the earth. A much smoother deceleration curve, with correspondingly less severe heating problems, can be expected. Moreover, the Martian atmosphere will prove less corrosive to structural materials.

When Mars is photographed through colored filters, several types of clouds show up. A blue or violet filter makes visible the general outline of the atmosphere. The reason for this is that the constituents render the atmosphere partly opaque in the violet spectrum, and thus the light is reflected, causing the plane to appear as having a larger diameter. The difference in diameter between unfiltered and filtered photographs is then equal to twice the depth of the atmosphere. On a "violet" photograph, the planetary disk appears to be spotted with clouds. These are presently thought to be density irregularities, altering the reflectivity, rather than actual cloud formations. They are often seen at an altitude of about 10 to 15

Table 1 Physical data of Mars

Distance from sun	
Mean	$141.5 \times 10^6$ miles
Aphelion	$154.1 \times 10^6$ miles
Perihelion	$128.0 \times 10^6$ miles
Distance from earth	
Perihelion opposition	$34.8 \times 10^6$ miles
Aphelion opposition	$61.5 \times 10^6$ miles
Aphelion conjunction	$340.0 \times 10^6$ miles
Orbital velocity	
Mean	14.98 miles/sec
At aphelion	13.64 miles/sec
At perihelion	16.45 miles/sec
Escape velocity	3.13 miles/sec
Circular velocity at surface	2.21 miles/sec
Equatorial diameter	4220 miles
Length of day	
Sidereal	$24^h 37^m 22^s.668$
Solar	$24^h 39^m 35^s.247$
Length of year	
Mars days	668.599
Earth days	686.979
Eccentricity of orbit	0.09336
Mean sidereal motion in 24 hr	1886.519 sec of arc
Inclination of orbit to ecliptic	$1^\circ 50' 59.8''$
Inclination of Martian equator to its orbit	$25^\circ 10'$
Heliocentric longitude of node (1956)	$49^\circ 13' 05.5''$
Heliocentric longitude of perihelion (1956)	$335^\circ 14' 56.6''$
Mass (Earth = 1)	0.108
Volume (Earth = 1)	0.151
Density (Earth = 1)	0.710
Density (Absolute)	3.910
Surface area (Earth = 1)	0.278
Gravity at surface (Earth = 1)	0.38

miles and are referred to as "violet clouds."

The "yellow" clouds—or those clouds visible on yellow filtered photographs—are seen at an altitude of about two or three miles. These are believed to be dust clouds, consisting of fine particles swirling up from the deserts. The dynamics of these particles is not known.



Drawing of Mars, based on visual observations from earth.

The existence of actual clouds of ice crystals or water droplets has not yet been proved, although a haze over the polar regions, visible only in the planet's autumn, may be found to be similar to our cirrus clouds.

### The Polar Caps

Two of the most striking features of the Red Planet are noted immediately by the observer. These are the brilliant white polar caps. The southern cap may extend as far as 4,000,000 sq miles at its maximum spread during the winter.

Temperatures below  $-100^{\circ}\text{C}$  have been measured for the south pole of Mars. The polar caps melt quickly in the summertime, a climatic condition which indicates the layers must be relatively thin, perhaps of the order of  $\frac{1}{2}$  in. The maximum temperature measured for Mars during the summertime at the equator was approximately  $20^{\circ}\text{C}$ . Temperatures as high as  $30^{\circ}\text{C}$  have been detected for dark, heat-absorbing areas. The mean temperature of Mars is  $-40^{\circ}\text{C}$  against  $15^{\circ}\text{C}$  for the earth.

Atmospheric pressure on Mars is sufficient for the subsistence of free water at  $40^{\circ}\text{C}$ , whereas the limited amount prohibits actual flow. Water propagation is believed to take place in the vapor phase. The rapid disappearance of the ice caps reveals highly irregular boundary lines, indicating surface temperature differences may be caused by slopes in the landscape. In 1956, the southern cap was observed to have melted completely.

During the fall, a haze forms over the polar region, obscuring vision until mid-winter, when the gleaming ice cap again becomes visible.

### The Dark Areas

Early observers designated the dark regions of Mars as "seas," an ascription influenced, no doubt, by the same considerations which gave rise to the "seas" of the moon. It is unlikely that large areas of free water exist on Mars. This is a generality based upon knowledge that water content of the atmosphere is extremely low. Also, if there were such a body of water, the sun would be reflected by it as a shiny image. So far no such image has been observed.

Another theory about the dark areas was presented in 1878 by a French astronomer.<sup>2</sup> Pointing out the seasonal change in color from bluish green to brown, and even yellow, he concluded that green plants similar to those common to the earth were present on Mars. Unhappily for this theory, the color changes do not follow the seasons as on earth. The greens of winter and spring turn blue in early summer and "wilt" as early as mid-summer, displaying brown, carmine, and maroon colors at the time of the Martian year when the temperature is at its peak.

Others<sup>3</sup> propose that Martian vegetation may be extremely resinous, since it is believed that this would allow the survival of growth under low pressure and temperature extremes. The resin would absorb the infrared light, utilizing it for heating purposes, leaving the reflected sunlight poor in this part of the spectrum.

The nonvegetation theories center on two concepts. The first was propounded by Arrhenius, who attributed the color changes to a hygroscopic mineral. Many

chemicals have a property which enables them to change color when the moisture content is varied. A common example is cobalt chloride, a pink salt which turns blue when dehydrated.

The moisture from the melting polar caps of Mars was once believed to produce a similar reaction in its minerals. The theory is now abandoned, since the latest examination of the atmosphere disproves the probability of significant amounts of water. Another contradictory feature is the color changes, which are too irregular to support this hypothesis.

The most recent theory deals with the existence of active volcanoes on Mars.<sup>4,5</sup> Elucidating his idea that the dark areas repeat themselves in shape and general direction around the globe, McLaughlin concluded that volcanic dust is carried up in the thin atmosphere and spread about by the principal winds. These winds are believed comparable to our trade winds and much effort has been put into proving the similarities.

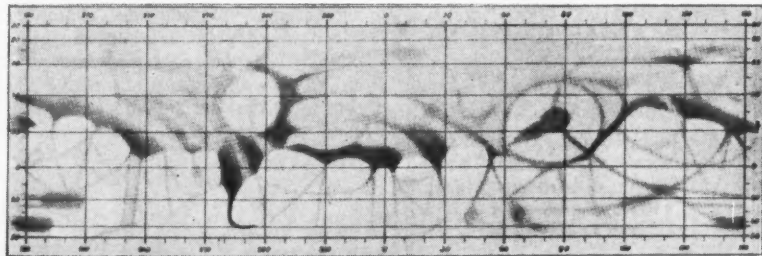
Although this theory has many drawbacks—it does not, for example, explain the appearance of the dark fringe areas around the receding polar caps—it is interesting. One may not, however, accept readily an assumption of the existence of volcanoes on a planet the size of Mars, with its very low gravity, its smooth surface, and its lack of shifting seas. Tidal effects must be very slight due to the planet's great distance from the sun and its possession of relatively small moons. These factors are predominant in the creation of volcanic action on the earth.

The most recent interpretation of the dark areas was presented by Kuiper in December 1956. He proposed that the dark regions were solidified lava. The consistency of this surface material would account for the fact that the dark areas always reappear quickly after severe duststorms. Winds could easily blow dust off such deposits, since they would be left with glassy surfaces following rapid solidification.

### The Bright Areas

Almost as mysterious as the dark areas are the bright, orange-colored desert

<sup>2</sup> Lias, E., et al., "Memoire de Mars."  
<sup>3</sup> Slater, A. E., "The Colours of Martian 'Vegetation,'" *Space Flight*, vol. I, no. 1, Oct. 1956, pp. 35.  
<sup>4</sup> McLaughlin, D. B., "Interpretation of Some Martian Features," *Publ. Astron. Soc. Pacific*, vol. 66, Aug. 1954, pp. 161-169.  
<sup>5</sup> McLaughlin, D. B., "Further Notes on Martian Features," *Ibid.*, vol. 66, Oct. 1954, pp. 221-228.



Map of the Red Planet, constructed from a number of photographs and drawings.

## ENGINEERS

*Aerodynamics & Propulsion*

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regions which give the planet its overall reddish appearance. It is now an established fact that these areas are completely devoid of any form of life. The compound mineral that covers the desert is believed to be felsite, an igneous rock of potassium and aluminum silicates with quartz inclusions, or limonite, a rustlike iron oxide. The elevation of the formations in the desert region does not at any point exceed 6000 to 9000 ft. Nix Olympica is thought to be the highest mountain on the planet.

#### The Canals

In strong magnifications, the desert regions are seen to be traversed and surrounded by faint, seemingly straight lines. These are the famous canals, which have remained an enigma for some 80 years. Not unlike a network of bluish veins, they were last observed by Richardson<sup>6</sup> during the most recent opposition. Although he did not succeed in photographing them—the images on the photographic plates were only  $\frac{1}{10}$  in. in diam—he could clearly distinguish them through the 60-in. telescope when using high magnification.

Many astronomers have seen and accepted the fact that Mars is scarred with what often appears to their eyes as highly symmetrical grooves. The nature of these grooves, in the meantime, is unfounded. Astronomers have fought bitterly over the origin and significance of the lines. Lowell and Barnard represented the contestants in this scientific tug-of-war. Lowell concluded that the Martian canals were built by intelligent beings who were struggling to maintain a water supply from the polar caps on the drying planet. Barnard, his strongest opponent, maintained the canals were natural formations.

The latest observations strongly support this hypothesis. Even though a true definition of the canals remains to be given, the "intelligent life" theory seems to have been wholly discarded. The only life we may expect to find on Mars will be that which is believed to constitute the dark areas. Even this, a very low form of life, would be capable of withstanding temperature extremes. Lichens are an outstanding example of symbiosis. Being formed by an alga and a fungus, the combination of these two plants, the lichen, is capable of living through environmental conditions which would prove fatal to any other plant. The alga contains chlorophyll and is thus the energy absorber. The fungus thrives on the alga and at the same time protects it from desiccation by providing it with water.

<sup>6</sup> Richardson, Robert L., "Preliminary Report on Observations of Mars at Mt. Wilson in 1956," *Spaceflight*, vol. I, no. 3, April 1957, p. 114.

#### Expedition to Mars

A single example of a space flight project with Mars as the goal is described in a résumé of a book, "The Exploration of Mars," by Wernher von Braun.<sup>1</sup> The project seems quite ambitious, since the expenditure of fuel reaches a total of half a million tons of propellant! This, however, is only one-tenth of the quantity needed in von Braun's earlier "Mars Project," the enormous saving being achieved by using two ships carrying a dozen men, instead of 10 ships carrying 70. The propellants are nitric acid and hydrazine.

Both ships are constructed in a 1075-mile orbit above the earth. As they are not meant to be landed, they are not streamlined, but are assembled by girders and aluminum members without skin or fuselage. Only the propellant tanks and living quarters are voluminous bodies.

One ship is a passenger-carrying vessel which will make the round trip. The other, the "cargo ship," carries the stores needed for the journey and the landing craft which will make the actual descent to the planet. Each ship weighs 1870 tons on departure from the earth orbit, with empty propellant tanks and surplus motors jettisoned en route.

The most impressive item of the expedition is the 177-ton glider for the descent to Mars when an orbit has been established around the planet at a height of 620 miles. It will carry nine of the expedition's 12 men, who will spend 400 days on Mars, a period determined by the length of time it takes for earth and Mars to get into the right position for the return journey.

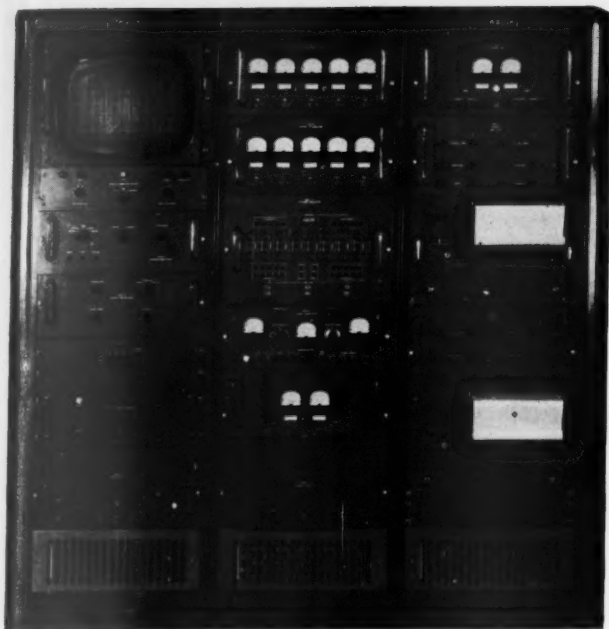
The landing party will need 18.7 tons of oxygen, water and food, and 35 tons of equipment, including two small tractors to allow a good radius of action. This explains the size of the glider, which also has to carry sufficient propellant to get the nine men and 5.5 tons of specimens back to the orbit.

When it is time to take off from Mars, the glider is stripped down and winched into the vertical position, becoming, in effect, a simple rocket which will be left orbiting Mars for the benefit of future expeditions.

The passenger ship then returns to earth. To economize use of propellants, it settles into an orbit at a height of 56,000 miles, from which crew and specimens are retrieved by a ferry rocket launched from the lower 1075-mile orbit.

Mars still hides many secrets from us. The project outlined here will be superseded by better ones, utilizing more powerful propulsion systems, and some time in the not too distant future Man will be able to see for himself whether or not his theories about the planet were correct.

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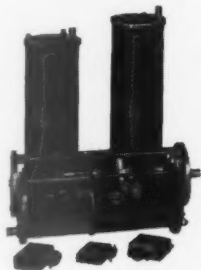
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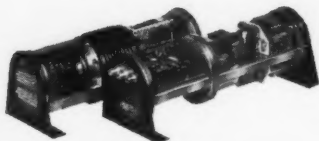


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# ARS News

## Hypervelocity Vehicles Take Shape At ARS Semi-Annual Meeting

**A** BLUNT-NOSED hypervelocity glide vehicle supported on re-entry by delta wings whose leading edges are also blunted was suggested at the ARS Semi-Annual Meeting in San Francisco earlier this month as the optimum design for a manned long-range craft.

Alfred J. Eggers Jr. of the National Advisory Committee for Aeronautics Ames Aeronautical Laboratory, Moffett Field, Calif., proposed the vehicle after exploring the favorable and unfavorable aspects of ballistic, glide and skip trajectories.

Dr. Eggers' paper, delivered at the Hypervelocity Flight Session, was one of 34 technical papers presented at eight sessions at the St. Francis Hotel June 10-13. About 1000 missile scientists and engineers attended the meeting.

Those three paths through the upper atmosphere before returning to earth would give a maximum range for a given velocity at the end of powered flight. Using this as a point of departure, Dr. Eggers pointed out that ballistic vehicles would be the least efficient, generally requiring the highest velocity at the end of powered flight. On the other hand, heating at re-entry can be kept low by making the nose blunt.

Glide vehicles, while far superior to ballistic counterparts if possessing a lift-drag ratio of four or greater, encounter severe heating problems. The energy can be radiated back to the atmosphere, however, he noted.

Skip vehicles of similar lift-drag ratios are comparable to the glide ship for range. But large aerodynamic loads and very severe heating during dips into the atmosphere make this type of trajectory the least attractive.

Extending this into manned craft, Dr. Eggers rejected the ballistic path because of the decelerations it would impose, and concluded that glide vehicles were the most promising.

Blunting the glider's leading edges would reduce heating, and to compensate for the drag penalty demanded by that configuration, the wings are swept to around 65 deg (see photo right).

Turning to satellite vehicles, and treating them as the limiting case of ballistic craft, he said that, with a tra-



Re-entry and recovery of a high-drag satellite.

jectory only slightly inclined to the horizontal, a finned hemisphere (10 ft diam, 1250 lb in weight, 16 lb per sq ft weight to frontal area ratio) would experience a maximum deceleration of 7.2 g and a maximum equilibrium surface temperature of around 2500 F. Ceramic outer surfaces coupled with radiation losses should permit integrity to be maintained for recovery (see photo above).

Heat transfer problems, as has been seen, determine not only the maximum speed which can be obtained but impose severe limitations on choice of flight

path and configuration of the vehicle. So stating, Jackson R. Stalder, also of NACA's Ames facility, outlined to the meeting the major difficulties encountered in the aerothermodynamic design of hypersonic craft.

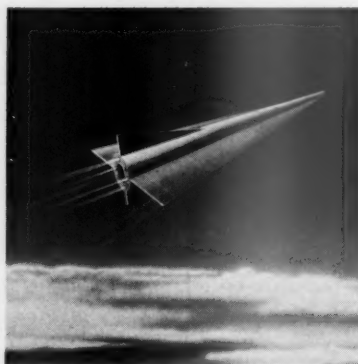
Because of the temperatures, continued Samuel A. Batdorf of Lockheed Aircraft, stress is no longer proportional to strain. Rather, a stress-strain-time-temperature relationship must be substituted, but when such a relationship is found, he warned, it may be too complicated to apply.

C. Frederick Hansen, a third NACA Ames man on the panel, dealt with some characteristics of the upper atmosphere pertaining to flight in the far supersonic ranges. And Lester Lees of CalTech rounded out the well-integrated session with a report on recent developments in hypersonic flow.

The hypervelocity session was chaired by Julian Allen of NACA's Ames Lab, who was recently honored for conceiving the blunt-nose design for the ICBM (see p. 726).

Besides hypervelocity vehicles, the sessions covered instrumentation and guidance, liquid rockets, solid rockets, combustion, ramjets and space flight.

The ARS Northern California Section sponsored the meeting. The Semi-Annual Meeting of the American So-



NACA's example of a hypervelocity glide vehicle.

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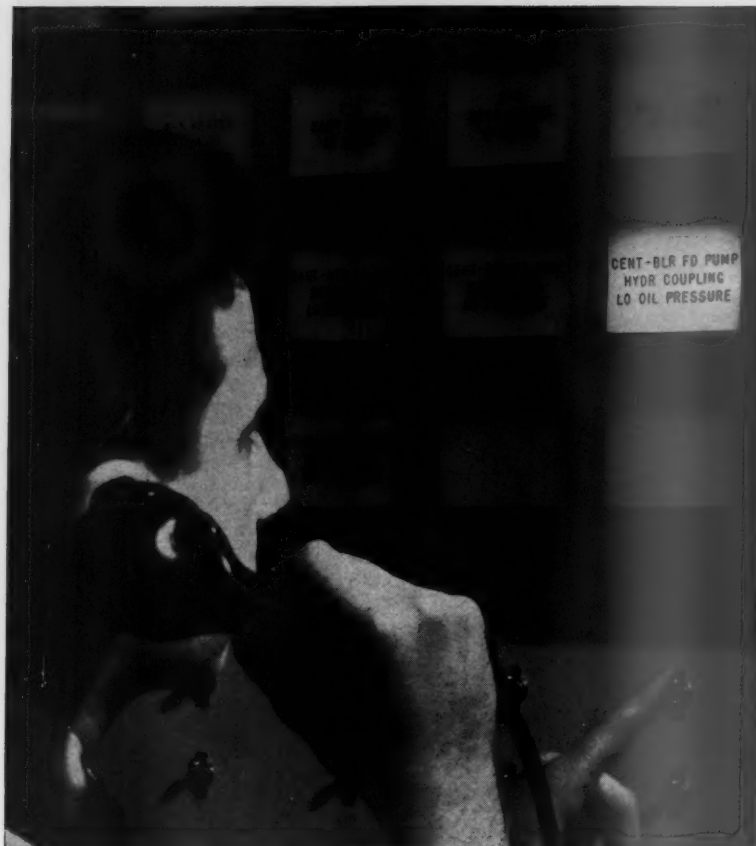
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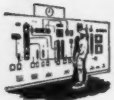
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ciety of Mechanical Engineers was held at the same time in the Sheraton Palace Hotel, and many members of both organizations sat in on sessions of both societies.

Dan A. Kimball, president of Aerojet-General, was the featured speaker at a luncheon on June 11, while Rear Adm. W. F. Raborn, director, Special Projects Office, Bureau of Naval Ordnance, was the principal speaker at the banquet on the evening of June 13 which closed out the meeting.

## ARS Adds Six Corporate Members

Six more firms active in the rocket or jet propulsion fields have become corporate members of the AMERICAN ROCKET SOCIETY. They are:

- Walter Kidde & Co., Inc., Belleville, N. J. The company is involved in research, development and manufacture of auxiliary power systems, high pressure containers and glass fiber radomes.

Representing the firm in ARS activities will be A. H. Hobelmann, assistant vice-president; J. R. Stanton, contract engineer; R. P. Kirkup and R. M. Belmonte, research and development engineers; and W. Masnik.

- Litton Industries, Inc., Beverly Hills, Calif. This company's activities are centered on development and production of electronic guidance and control equipment.

Designated as ARS representatives are H. W. Jamieson, vice-president; H. E. Singleton, director, guidance and control; R. A. Roche, director of research; G. F. Steele, director of computer research; and J. Ray Donohue Jr., director of military relations.

- The Meyercoed Co., Chicago, developers and manufacturers of high temperature film markings and interior and exterior nameplates, emblems and film markings for rockets, missiles, jet aircraft and components thereof.

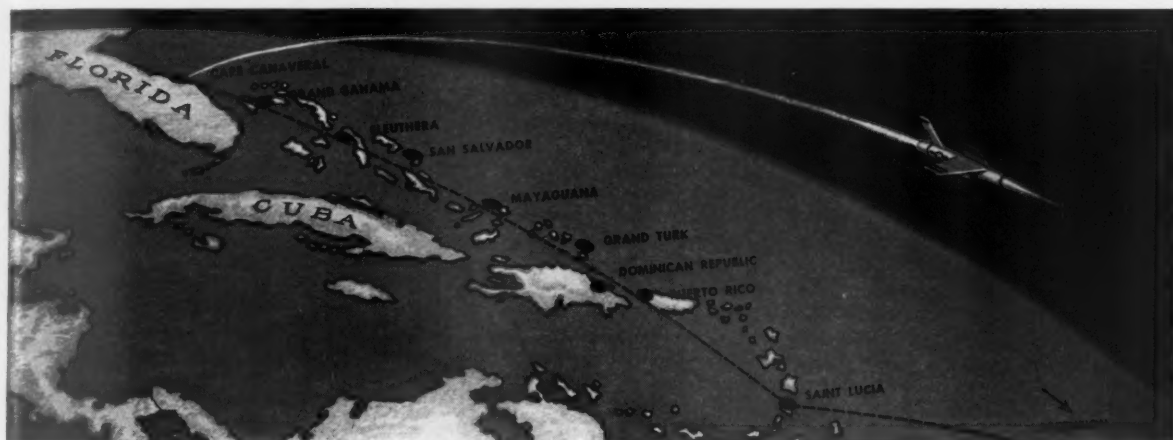
ARS representatives for the company are Leonard A. Knopf, vice-president; Ludwig F. Akkeron, chief chemist; D. J. Stockburger, manager, Nameplate Div.; John A. Cargill, manager, and E. R. Rodgers, assistant manager, Aircraft Marking Component Div.

- Norris-Thermador Corp., Los Angeles, producers of metal rocket components such as warheads, motors, nozzles and fins, and producers of JATO bodies.

Representing the firm in ARS will be Donald P. White, vice-president; Larry Shiller, chief research and development engineer; Charles E. Fisher and Fred A. Wheeler, ordnance engineers; and Turner C. Hawes, development engineer.

- Northrop Aircraft, Inc., Hawthorne, Calif., whose activities include research,

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These same techniques will also be used in the Vanguard Project to track the earth satellite when it is launched from Cape Canaveral.

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development and design of guided missiles.

ARS representatives are George N. Mangurian, chief analytical engineer; Ralph C. Hakes, preliminary design project engineer; Raymond E. Grill, preliminary design engineering specialist; Bernard B. Levitt, propulsion systems analysis supervisor; and Robert L. Nathan, propulsion systems analysis engineer.

• Telefilm, Inc., Hollywood, Calif., whose activities include technical motion picture servicing of classified films made by Southern California companies holding government contracts for the development and manufacture of rockets, guided missiles, electronic elements and aircraft.

Designated ARS representatives are Joseph A. Thomas, president; James B. Pinkham, executive vice-president; Peter Comandini, secretary-treasurer; Steven A. Guy, director, electronics department; and Mervin J. Updegraff, supervisor, art department.

### Pacific Rocket Society Holds Public Launching

The Pacific Rocket Society last month held its fourth annual public

launching at the Mojave Test Area in Kern County, Calif.

Dan Brewer, Guided Missile Research Div., Ramo-Wooldridge Corp., and vice-president of the Southern California Section of the AMERICAN ROCKET SOCIETY, was the guest speaker at a recent meeting of the amateur rocket group.

### Rocket Research Institute To Fire Mail Rockets

The Rocket Research Institute, an amateur experimental organization with headquarters in Glendale, Calif., will fire five mail-carrying rockets carrying a total of 5000 special covers across the Nevada-California border July 1. This second interstate rocket mail flight by the Institute will commemorate the start of the International Geophysical Year and marks the fourth such rocket-post demonstration in ten years.

Rockets to be used in the demonstration are 11 ft long and 3 in. in diam, and use a micrograin solid propellant developed by the Institute in 1943. Each rocket will carry 1000 covers approximately 1½ miles from a launching tower in Douglas County, Nev., to Mono County, Calif.

George S. James, Aerojet-General

Corp., Institute director, emphasized that the flight is not sponsored by the U. S. Post Office but that these rocket covers have the same status as covers carried in aircraft prior to the establishment of official U. S. Government air-mail service.

The purpose of the flight is to raise funds for an educational rocket engineering project being conducted in Sacramento by the Institute. This program, called SPARK I (Special Project, Altitude Rocket Knowledge), is aimed at the construction of a two-stage intermediate-altitude sounding rocket for launching during IGY 1957-1958. The two rockets for the SPARK I system are a 400-lb-thrust, 74-sec-duration, lox-alcohol sustainer and a 10,000-lb-thrust, 1-sec-duration booster.

### IAF Congress Slated For Barcelona

The Eighth International Astronautical Congress will be held Oct. 7-12, 1957, in Barcelona, Spain.

The largest and most comprehensive program yet scheduled by the International Astronautical Federation is now in preparation.

ARS papers being assembled by Space Flight Committee Chairman Krafft Ehrlicke cover a broad range of subjects. They include, tentatively, papers by Wolfgang Klemperer and E. T. Benedict of Douglas Aircraft; E. R. G. Eckert, University of Minnesota; David G. Simons, Holloman Air Development Center; C. Gazley and D. J. Masson, Rand Corp.; R. E. Roberson, Autonetics Div. of North American Aviation; R. Haviland, General Electric Co.; P. E. Glasser, Arthur D. Little, Inc.; R. T. Patterson, Martin Co.; H. H. Koelle, Army Ballistic Missile Agency; S. F. Singer, University of Maryland; Angelo Miele, Purdue University; and Mr. Ehrlicke.

Additional program and reservation information was recently mailed to all ARS members.

### Sections

**Alabama:** James J. Harford, ARS executive secretary, addressed the April gathering on the future aims and goals of the SOCIETY. A film on the V-2 was shown.

**Chicago:** Charles C. Miesse of the Armour Research Foundation spoke to the Section at the April meeting on the use of theoretical models in missile research. Mr. Miesse, who is supervisor of combustion research for the foundation, gave examples where such models have proved very useful.

**Columbus:** John Townsend Jr., head of the Rocket Sonde Branch



### 'Moonwatch' Practice Begins in Southwest

Phoenix, Arizona, citizens prepare for "Operation Moonwatch" atop a 12-story building in what is reportedly the first earth satellite observation post built and equipped with private funds.

Underwritten by a civic-minded

banker, the station cost several thousand dollars. It is equipped with a 24-ft T-mast, a dozen wide-angle telescopes for as many sighting benches, a short wave radio, tape recorder, and associated items.

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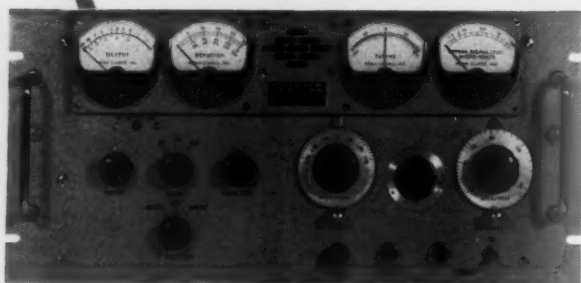
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NEMS-CLARKE special purpose receivers are designed to provide optimum performance for applications such as telemetering, guided-missile monitoring, radiosonde reception and numerous other applications where receivers of superior performance with high sensitivity and low noise are required.



## TYPE 1401-A RECEIVER SPECIFICATIONS

Type of Reception	FM/FM and PWM/FM
Frequency Range	216-245 Megacycles determined by plug-in crystals.
Noise Figure	Less than 7 db.
IF Bandwidth	Wide band—500 KC bandwidth at 3 db points. Attenuation $\pm$ 60 KC from center frequency greater than 60 db. Narrow band—100 KC bandwidth at 3 db points. Attenuation $\pm$ 250 KC from center frequency greater than 60 db.
Video Output	Sensitivity—0.16 volts peak-to-peak per KC of deviation. Frequency response within 3 db. AC coupled—10 CPS to 100 KC per second. Adjustable output control on front panel.
VU Meter in Video Output Circuit	Frequency response, flat over frequency range of 400 cycles to 80,000 cycles. Provided with front panel adjustable reference level control.
Spectrum Display Output	Provisions for connecting a 30 MC Spectrum Display Unit (NEMS-CLARKE Inc., SDU-200)
Frequency Monitor Output	30 MC
Frequency Deviation Meter	Peak reading over frequency range from 400 to 80,000 CPS. Three scales 25, 75 and 150 KC.

## NEMS-CLARKE INCORPORATED

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of the Naval Research Laboratory, spoke at the May dinner meeting on the earth satellite and rocket research. At the previous meeting, Ralph Bloom and Frank Kraft of Becco Chemical talked about concentrated hydrogen peroxide, its characteristics and uses.

M. W. Jack Bell has been appointed Section representative to the Columbus Technical Council and Harold Rienstra has been appointed representative to the Youth Guidance Committee. Three committees have been set up: A program committee with Loren Bollinger, William Weber and Thomas Kirk; a by-law committee with Donald Hall, Alex Lemmon and Albert Weller; and a publicity committee with Mr. Bell and Arthur Greshemer.

**Detroit:** Kurt R. Stehling of the vehicles branch, Project Vanguard, NRL, was the speaker for May, describing space investigation by use of high altitude rockets and space satellites.

At an earlier meeting, Donald R. Green, head of the engineering and instrumentation department, Naval Test Facility, White Sands Proving Grounds, spoke on "Rocket Test Facilities Operations." The Section also toured the Aeronautical Engineering Department, Aircraft Propulsion Laboratory and the Ford reactor at the University of Michigan.

**Ft. Wayne:** Alfred J. Zaehring, president, American Rocket Co., Wyandotte, Mich., was the guest speaker at the second anniversary banquet of the Ft. Wayne Section. Mr. Zaehring's topic was "Soviet Missiles."

**Maryland:** At its April meeting, the Maryland Section presented its first annual award to Grey R. Brooks, Thiokol Chemical Corp., Elkton Div., Elkton, Md. Mr. Brooks was honored as the engineer who had contributed the most to the advancement of the science of rocketry in the Maryland area for the year 1956. The award committee, headed by Joel Jacobson, vice-president of Aircraft Armaments Inc., chose Mr. Brooks from a list of over 50 candidates presented by the various companies and organizations in the Maryland area.

Mr. Brooks was selected for the award for his work on the Cajun rocket, which demonstrated his "keen understanding of rocket design, interior ballistics, and his ability to achieve high engine performance in a minimum period of time." The award was presented by Ivan Tuhy of The Martin Co. (photo) at a dinner meeting attended by more than 100 members and their wives.

Following the presentation of the award, R. T. Patterson, project engi-

## IMPORTANT DEVELOPMENTS AT JPL



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*The Jet Propulsion Laboratory is a stable research and development center located to the north of Pasadena in the foothills of the San Gabriel mountains. Covering an area of 80 acres and employing 1550 people, it is close to attractive residential areas.*

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For many years the Jet Propulsion Laboratory has pioneered in the design and development of highly accurate missile guidance systems, utilizing the most advanced types of gyroscopes, accelerometers and other precision electro-mechanical devices. These supply the reference information necessary to achieve the hitherto unattainable target accuracies sought today.

The eminent success of the early "Corporal" missile flights shortly after World War II firmly established the Laboratory as a leader in the field of missile guidance. These flights also initiated experiments involving both inertial and radio-command systems employing new concepts of radar communication. Because of this research and experimentation JPL has been able to add materially to the fund of knowledge

available to designers of complex missile systems.

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## Maryland's Missile Engineer of the Year



Grey R. Brooks of the Thiokol Chemical Corp. is presented with the Maryland Section's award as the area's outstanding engineer of the year by Ivan Tuhy (left) of The Martin Co.

neer, New Design, Vanguard Project, Martin Co., gave a talk on "Recovery of Scientific Research Payloads from High Altitudes." This talk reported on the results of a study by Martin personnel to determine the feasibility of the physical recovery of scientific-research payloads (such as nuclear emulsions exposed to cosmic rays; animal tissue, structural materials and equipment exposed to radiation; and geodetic, weather, astronomic and spectroscopic research equipment) from very high altitude, near-vertical trajectories.

After discussion of the magnitude of the thermal problem and the deceleration problem as the payload enters the earth's atmosphere, as well as the problem of locating the landed payload, it was concluded that, through approximate design considerations, such recovery is feasible today. The talk was illustrated with well-chosen lantern slides.

**New England:** Albert Clark of General Electric's Missile and Ordnance Systems Dept., Philadelphia, predicted a missile on the moon with instrumentation and transmitting equipment in the next 20 years. He spoke to a combined meeting of the ARS Section and the IAS group in Boston.

Members and guests of the New England Section last month visited the National Northern Corp., West Hanover, Mass., for a plant tour that included a view of JATO igniter production facilities and static and dynamic firings of various kinds of

projectiles and warheads, as well as gas generators, tracers and propellant systems.

**New York:** A discussion of frozen free radicals and their potentialities as propulsion fuels by Herbert P. Broida, Free Radicals Section Chief, National Bureau of Standards, highlighted the May meeting of the New York Section. Dr. Broida, now coordinating a major research effort by the Bureau in this field, discussed techniques now being used for the study of free radicals, illustrating his talk with a short movie and slides.

**Northern California:** Joseph Kaplan of the University of California at Los Angeles, Chairman of the U. S. National Committee for the International Geophysical Year, was the guest speaker at a joint meeting of the Northern California Section and the IAS last month. Dr. Kaplan discussed plans for IGY, with particular attention to Project Vanguard.

**Princeton:** The group played host to 150 visitors, including many from New York and Philadelphia, May 4 for an ARS Open House and tour of the Forrestal Research Center. Facilities of the University's Department of Aeronautical Engineering were open to the group. Highlight of the tour was a series of bipropellant rocket runs.

**San Diego:** S. A. Schaaf, associate professor of engineering and director of the low pressure research project at the University of California, was the guest speaker at a recent joint meeting of the San Diego Section and the IAS.

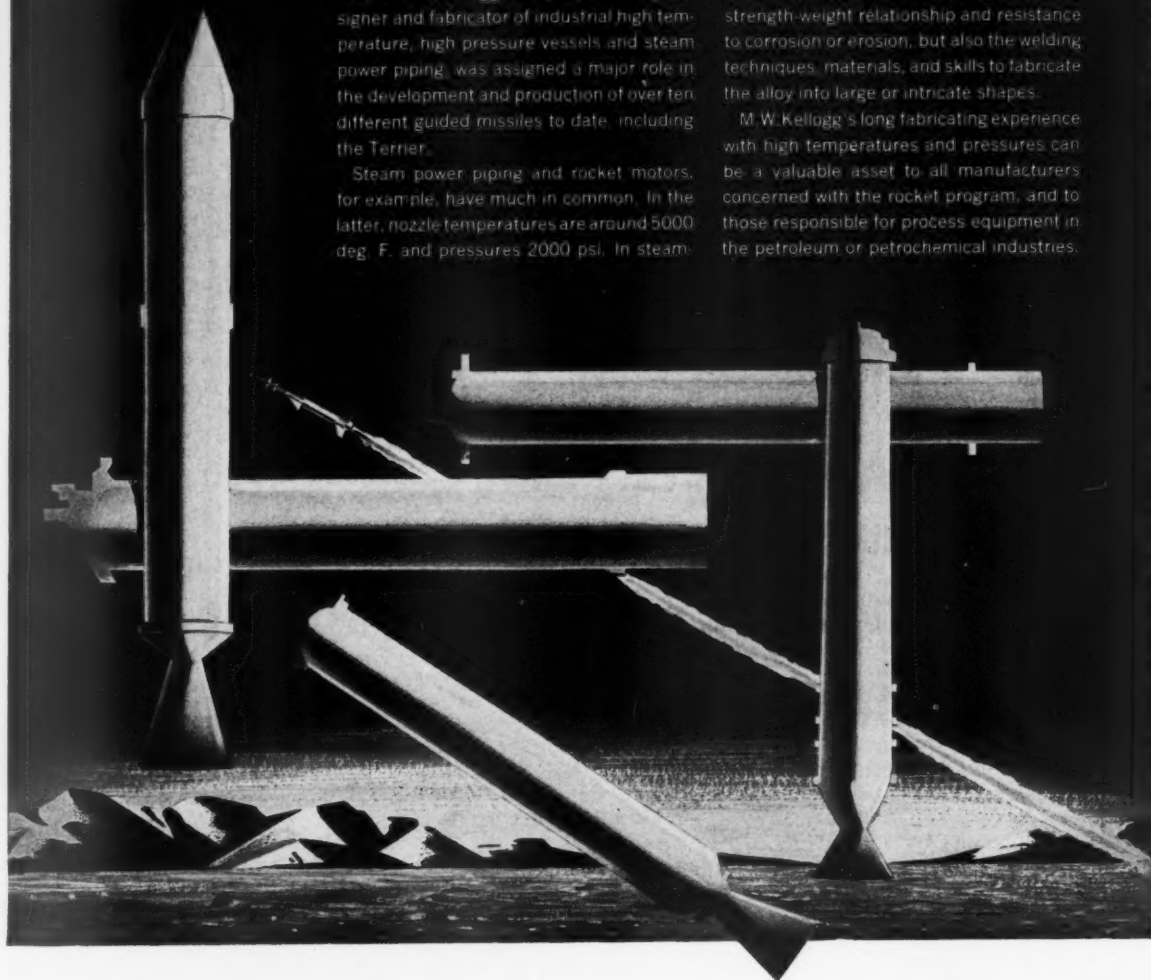
## Outline of 1000 Temperature-Pressure Problems

Any rocket engine looks simple in silhouette—but its 150 or more parts present new and complex problems of operating temperatures and pressures. This is the reason why The M. W. Kellogg Company, leading designer and fabricator of industrial high temperature, high pressure vessels and steam power piping, was assigned a major role in the development and production of over ten different guided missiles to date, including the Terrier.

Steam power piping and rocket motors, for example, have much in common. In the latter, nozzle temperatures are around 5000 deg. F. and pressures 2000 psi. In steam-

electric power plants, the Kellogg assignment concerns piping to withstand 1250 deg. F. and pressures over 5000 psi. Both problems include selection or development of not only the proper alloy to provide correct strength-weight relationship and resistance to corrosion or erosion, but also the welding techniques, materials, and skills to fabricate the alloy into large or intricate shapes.

M. W. Kellogg's long fabricating experience with high temperatures and pressures can be a valuable asset to all manufacturers concerned with the rocket program, and to those responsible for process equipment in the petroleum or petrochemical industries.



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Jet engine exhaust deflector manufactured by All American Engineering Company showing DuPont Aircraft Rivets used in its assembly.

## How Du Pont Aircraft Rivets solve high-temperature fastening problems

Like many other companies now doing research and development work in the higher temperature range, the All American Engineering Company of Wilmington, Delaware, uses DuPont Aircraft Rivets when blind fastening is required.

The A-286 Superalloy Rivet retains high shear and tensile strength even when subjected to extreme vibration at 1200°F., as it is in this jet engine exhaust deflector. These rivets must withstand continuous load reversal and stress cycles without failure.

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DuPont Aircraft Rivets for high-temperature applications are manufactured with either modified brazier or 100° countersunk heads in A-286 Superalloy and nickel. Currently available in three nominal diameters,  $\frac{1}{8}$ ",  $\frac{5}{32}$ ",  $\frac{3}{16}$ ", and a wide variety of lengths.

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Dr. Schaaf, one of the earliest researchers in low density, high speed gas dynamics, spoke on "Rarefied Gas Aerodynamics Research," reviewing work done at the Berkeley low pressure wind tunnel and using molecular beam apparatus. Approximately 100 people attended the meeting.

**Southern California:** Kraft Ehricke, assistant to the technical director, Convair-Astronautics, was the guest speaker at the May 9 meeting of the Section, attended by more than 250 members and guests. Mr. Ehricke's topic was "Space Flight Mechanics as related to Space Vehicle Propulsion Systems." He also discussed the effect of gravitational perturbances on satellite vehicles in cislunar, translunar and interplanetary space.

Irwin Hersey, ARS Director of Publications, was a guest at the meeting and spoke briefly on ASTRONAUTICS, the new ARS magazine slated for publication this summer.

This month's meeting will feature a talk by Paul Winternitz, Chemistry Dept., New York University. Scheduled for June 13 at the Rodger Young Auditorium in Los Angeles, the address will cover jet reactors, past and present.

**Twin Cities:** John E. Barkley has been named membership chairman and Carl L. Kober program chairman of the Section. Both men are in General Mills' Mechanical Div.

### Student Chapter News

**Polytechnic Institute of Brooklyn:** Willy Ley spoke to the Chapter at the May meeting on fuels and their application to space flight.

At an earlier meeting, Dominic B. Edelen, senior engineer-structures, Project Vanguard, Martin Co., spoke on the launching and problems encountered in Project Vanguard.

The newly elected officers, who will take office in September, are Paul Cooper, president; Thomas Lardner, vice-president; John Boccio, secretary; and Fred Schuyler, treasurer.

### ARS Meetings Calendar

**Aug. 25-28:** ARS-Northwestern Technological Institute Gas Dynamics Symposium, Northwestern University, Evanston, Ill.

**Dec. 2-6:** ARS Twelfth Annual Meeting, Hotel Statler, New York.

**Dec. 6-7:** ARS Eastern Regional Student Conference (under the auspices of Polytechnic Institute of Brooklyn Student Chapter), Hotel Statler, New York.



# Sunday Punch!

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# People in the News

## APPOINTMENTS

• **Frank Pace Jr.**, former Secretary of the Army, has been elected president of General Dynamics Corp., succeeding the late John Jay Hopkins. Pace had been executive vice-president and vice-chairman of the board since 1953. In addition, **Earl Johnson** has been named executive vice-president of the corporation. He was previously a member of the board and vice-chairman of the board of management.

• **Kraft Ehricke** has been named assistant to the technical director, Convair-Astronautics, and will head up a new preliminary design section for the division. He was previously chief, preliminary design section, systems analysis.



Beehan

McKenzie

• **T. Edward Beehan** has been elected secretary and **Reginald I. McKenzie** treasurer of Aerojet-General Corp. Other changes in company personnel include **Chandler C. Ross**, manager, Liquid Engine Div., to vice-president-engineering; **Robert B. Young**, resident manager, liquid rocket plant, Sacramento, and a member of the technical directorate, to vice-president, liquid rocket plant; **Robert J. Mill**, resident manager, metal parts manufacturing, to manufacturing manager, liquid rocket plant.



Ross

Young

• Two appointments in the research and development branch, Missile Systems Div., Lockheed Aircraft Corp., are **Ronald Smelt**, former chief of wind tunnel operations for ARO, Inc., and one-time head of guided missiles for all the British armed

forces, named first director of the new design office; and **Joachim Muehlner**, appointed consulting scientist. Dr. Muehlner was previously technical director of the Range Instrumentation Development Div., Army Ordnance, White Sands Proving Ground.

• **Howard W. Merrill** has been named a vice-president of The Martin Co. and general manager of the Baltimore Div., where he was formerly director of operations. Mr. Merrill succeeds **William B. Bergen** who will retain his position as executive vice-president.

• The Armour Research Foundation has named **Frank Genevese** an assistant manager of the propulsion and fluid mechanics research department. Before joining Armour, Dr. Genevese was at the Institute for Defense Analysis.

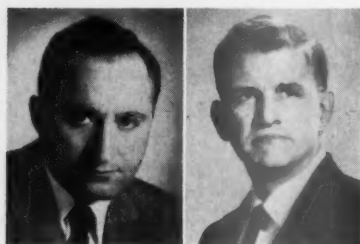
• **J. M. Miller** and **E. F. Lapham** have been appointed to key design posts in the missiles section, Bendix Products Div., Bendix Aviation Corp. Mr. Miller will be responsible for equipment and Mr. Lapham, as chief electrical engineer, for electronic systems for the Talos. At the same time, **Theodore Voorhees** has been appointed manager of Bendix' International Div.



J. M. Miller

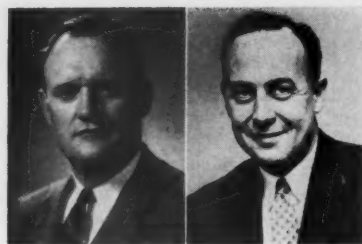
Lapham

• **Eugene Miller** has been appointed director of research and development of solid propellants in the Explosives Div., Olin Mathieson Chemical Corp. Dr. Miller was formerly chief of research laboratories engaged in studies related to guided missiles and rockets at Redstone Arsenal.



E. Miller

Rankin



J. C. Smith, Jr.

White

• Three major promotions in Chrysler Corp.'s missile operations are those of **John P. Butterfield**, to executive engineer in charge of all company missile operations; **Mansel F. Rankin**, to manager of missile operations at Huntsville; and **James C. Smith Jr.** to chief engineer for product planning, Detroit. Mr. Butterfield was formerly chief missiles engineer and Mr. Smith was chief engineer for marine missile systems with Chrysler.



J. R. Smith

Kerr

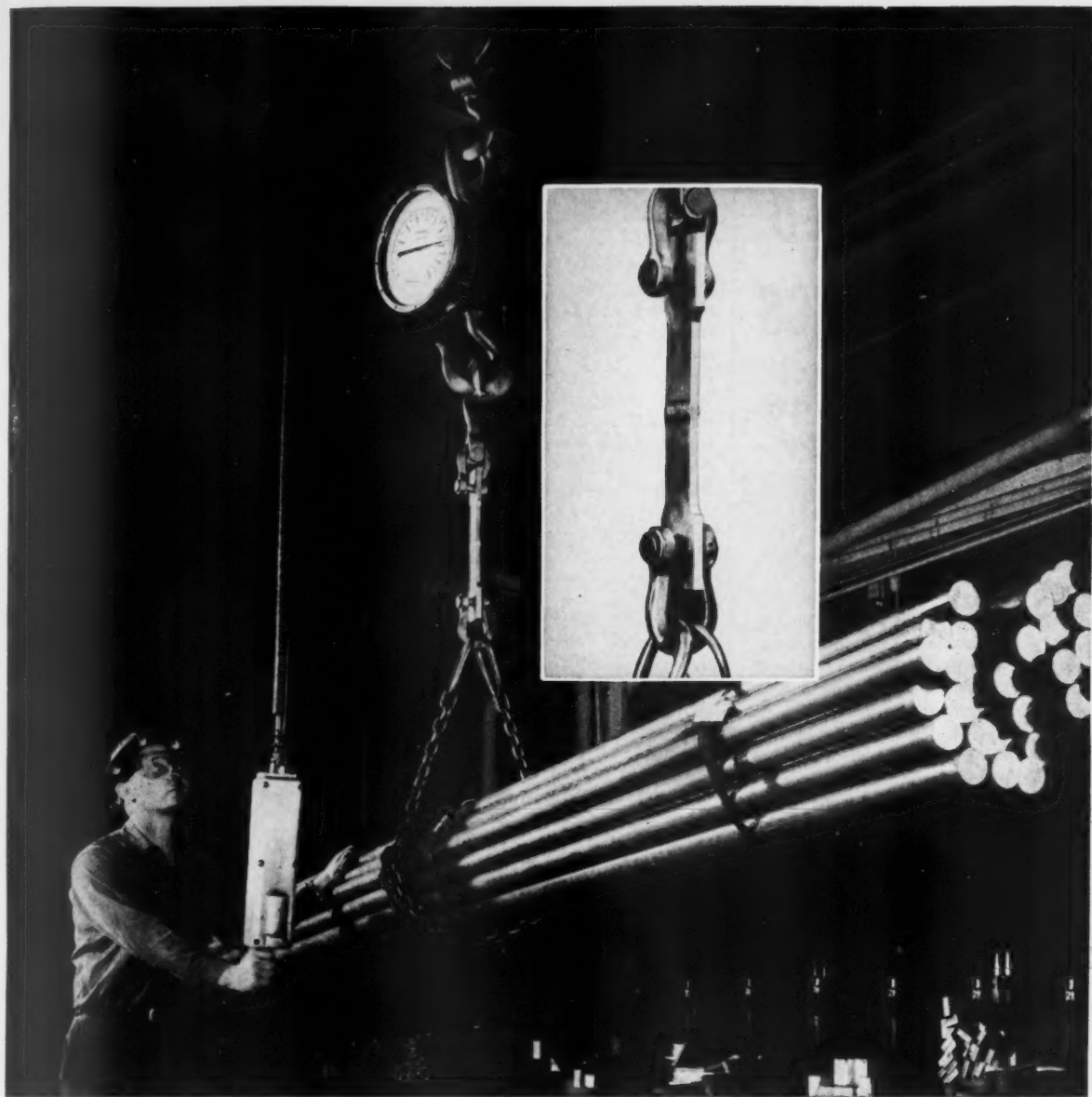
• **Perry A. White**, general controller of Baldwin-Lima-Hamilton Corp., has been named to the board of directors to succeed **Marvin W. Smith** who is retiring as chairman of the executive committee. The company's Standard Steel Works Div. presently includes a rockets and missiles department for crash basis production and delivery of rocket and missile parts. **J. Richard Smith**, who has been with the company for 20 years, will head the new department.

• Atlantic Research Corp. has gained two new staff members in the field of solid propellant rocketry. They are **Robert S. Scheffee**, formerly at The Martin Co., and **Harold W. Gear**, previously at Magnavox Corp., who have joined the Chemical Engineering Div.

• **John W. Anderson** has been named director of engineering for the inertial guidance facility being erected near St. Petersburg, Fla., by Minneapolis-Honeywell Regulator Co. Anderson was a chief engineer at the main Aeronautical Div. **Charles H. Kelly** has been named personnel manager for the new operation.

• Avco Manufacturing Corp. has named **James R. Kerr**, an Avco vice-president, president of the Lycoming Div.

JET PROPULSION



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Nichols

• **John M. Olin**, chairman of the board and of the executive committee since the formation of the Olin Mathieson Chemical Corp. in 1954, has become chairman of the financial and operating policy committee, while retaining his position of chairman of the executive committee. **Thomas S. Nichols**, president of the corporation since 1954, has been named chairman of the board of directors and **Stanley de J. Osborne** will move up to the office of president. **John W. Hanes**, a company director and former chairman of the finance committee, has reached retirement age but will serve as financial consultant and on the financial committee.



de J. Osborne



Hanes

• The Lockheed Georgia Div. of Lockheed Aircraft Co. has promoted **Roy Mackenzie**, former assistant manufacturing manager and co-developer of the Unitwin airplane, to manufacturing manager; **William B. Rieke**, former production manager, to assistant manufacturing manager; and **William A. Benson** to assistant production manager. **Ralph Osborn**, former manufacturing manager, has moved up to the position of executive vice-president of Lockheed Aircraft Services, Inc.

• **W. R. Miller** has organized the firm of Miller Associates, 742 South Hill St., Los Angeles, which will specialize as management consultants in industrial, electronic and technical executive procurement. Previously, Mr. Miller was vice-president in charge of manufacturing, Longren Aircraft Co., and has been with Rheem Manufacturing, Convair and TWA.

• **Jacob L. Zar** has been appointed director of engineering of the Air-

borne Accessories Corp. Prior to joining the company, he worked in the Guided Missiles Div., Republic Aviation Corp.

• **Specialties, Inc.**, has named **Heinz Fornoff** vice-president for engineering. Prior to this, Fornoff was department head of drone missile flight control, Sperry Gyroscope Div., Sperry Rand Corp.



Zar



Fornoff

• **Vernon I. Weihe**, formerly technical assistant to the vice-president in charge of engineering at Melpar, Inc., has joined the Avionic Div., General Precision Lab., Inc., to direct the avionic systems planning activities from Washington.



Weihe



Bolton

• **James A. Bolton**, former acting manager of the battery unit, Raleigh laboratory, American Machine and Foundry Co., has been promoted to manager of the laboratory, succeeding **Haywood C. Smith**, who has been appointed technical manager, research and development department, Engineering Div.



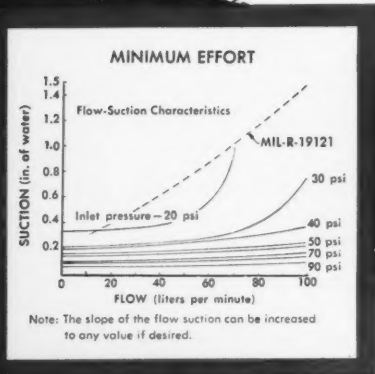
Teree



Lou

• **Baboo Ram Teree**, chief engineer and manager of engineering and manufacturing, Greer Hydraulics, Inc., has been elected vice-president in charge of engineering and manufacturing.

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• **Seymour B. Cohen** has been appointed manager of the antenna systems laboratory, Stanford Research Institute. Dr. Cohen was formerly head of the microwave group.

• **Nils H. Lou**, former works manager, Republic Aviation Corp., has joined Reynolds Metals Co. as manager of manufacturing, Parts Div.

• **Norman C. Anderson**, manager of the Photoconductor-Transistor Div., Electronics Corp. of America, has been promoted to division vice-president.

• **Kellett Aircraft Corp.** has named **Robert P. Williams Jr.** its Washington representative. He was formerly head of Rheem Manufacturing Co.'s Washington office.

• **James A. Broadston** has been named manager of the newly formed Rocketdyne Service Div., which will provide logistic support to users of Rocketdyne engines. Mr. Broadston was formerly manager of Rocketdyne's propulsion field laboratory at Santa Susana, Calif.

#### HONORS

• **H. Julian Allen**, chief, High-Speed Research Div., Ames Aeronautical Laboratory, last month received the NACA's highest honor, the Dis-

tinguished Service Medal, for his work on ballistic missile nose cones. Mr. Allen was honored for his discovery five years ago that a blunt nose cone would be more effective than a sharp-nosed cone in dissipating heat energy developed upon re-entry into the earth's atmosphere.

• Thirty new members have been elected to the National Academy of Science, including **Hendrik W. Bode**, director of research in physical sciences, Bell Telephone Laboratories; **Charles S. Draper**, professor of aeronautical engineering, MIT; **Jesse L. Greenstein**, staff member, Mount Wilson and Palomar Observatories; **Joseph Kaplan**, professor of physics, University of California and chairman, USNC-IGY; **Howard J. Lucas**, emeritus professor of chemistry, California Institute of Technology; **Jerrold R. Zacharias**, professor of physics, MIT.

• **Willis M. Hawkins**, assistant general manager of Lockheed Missile Systems Div., has been appointed a member of the U. S. Army Scientific Advisory Panel which advises the Secretary of the Army, Chief of Staff and Chief of Research and Development on scientific and related matters.

• **Harry F. Guggenheim**, president of the Daniel and Florence Guggenheim Foundation, has received the Laura Taber Barbour Award, created to recognize notable achievements in the promotion of safety in air flight.

• **Alan T. Waterman**, director of the National Science Foundation, has been awarded the first annual Conrad Award, established by the Office of Naval Research, for his work as a leader in the development of ONR in government-science relationships which resulted in an expansion of scientific support by other government agencies.

#### DEATHS

• **Oscar P. Hass**, director of flight operation, Republic Aviation Corp., died recently. He was one of the nation's top test pilots.

• **John Jay Hopkins**, chairman of the board of General Dynamics Corp., which built the USS Nautilus, died last month. He was also chairman of the board and managing director of Canadair, Ltd., a General Dynamics subsidiary, as well as organizer of the General Atomic Div. of the company.

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# New Patents

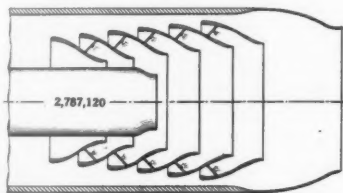
George F. McLaughlin, Contributor

**Thermal shock-resistant ceramic body (2,785,080).** Floyd A. Hummel, State College, Pa., assignor to The Carborundum Co.

Body made by forming finely divided particles of  $\text{Li}_2\text{CO}_3$ ,  $\text{Al}_2\text{O}_3$  and  $\text{SiO}_2$  which yield upon firing a crystalline structure composed of  $\text{Li}_2\text{O}$ ,  $\text{Al}_2\text{O}_3$  and  $\text{SiO}_2$  in the range of 1:1.2 to 1:1.10 of solid oxides, and sintering the body at a temperature between about 1000 C and the liquidus temperature of the mass between 1300 and 1450 C.

**Apparatus for utilizing the earth's magnetic field to indicate aircraft velocity (2,785,376).** Alan Hazeltine, Maplewood, N. J., assignor to Hazeltine Research, Inc.

Conductor elements are rotated about a horizontal axis for periodically reversing the polarity of the voltage along the elements as they are moved through the earth's magnetic field. Alternately, signals of maximum and minimum magnitude are jointly representative of the true ground velocity of the aircraft.

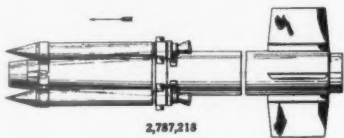


**Plural annular coaxial combustion chambers (2,787,120).** Rene Leduc, Argenteuil, France.

Frustoconic blades in an annular duct, of a diameter which decreases at a greater rate at the smaller end of the duct than the diameter of the blades in the larger end of the duct. The annular passages contain a plurality of combustion means.

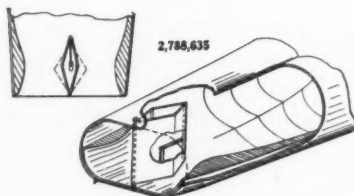
**Gun installation in jet aircraft (2,787,194).** Paul F. Peterson, Palos Verdes Estates, Calif., assignor to North American Aviation, Inc.

A baffle spaced forwardly of the gun sufficiently to encounter expanding discharge gases from the gun, deflecting the gases away from the engine air inlet.



**Aircraft (2,787,218).** A. Anthony, Farnham, England, assignor to the Government of Great Britain.

Device to jettison booster propulsion elements from aircraft, comprising a frangible bolt securing an anchoring member to the aircraft body.



**Device for varying a jet engine orifice (2,788,635).** Frederick A. Ford Jr., Whittier, Calif., assignor to North American Aviation, Inc.

A collapsible wedge vertically mounted in the nozzle throat and adapted for movement so as to vary its angle. Movement of the wedge inversely varies the cross-sectional area of the throat.

**Locking arrangement for singly and sequentially releasing ballistic missiles (2,788,712).** Nils-Erik Gustaf Kuller and Karl-John Thorild, Bofors, Sweden, assignors to Aktiebolaget Bofors Corp.

A cluster suspended from the structure of an aircraft in which all the missiles are joined slidably relative to each other. Each missile co-acts with the locking member of the next higher missile to retain the member in locked position, the locking member of the lowermost missile being free to move into its release position.

**Tail cone (2,788,803).** Joseph R. Greene, San Diego, Calif., assignor to Solar Aircraft Co.

Metal pressings providing in one piece a part of the annular wall which connects an adjacent pair of struts extending outwardly across the fluid passage. Integral joints interconnect the pressings.

**Cooling of turbine rotors (2,788,951).** John A. Flint, Farnborough, England, assignor to Power Jets (Research and Development) Ltd.

Three-stage axial flow gas turbine with a hollow driving shaft through which coolant fluid is directed to the chambers in the middle of the successive series of separate chambers.

**Wind tunnel force and moment measuring device (2,785,569).** Nairn L. Miller, Ogdensburg, N. Y., assignor to North American Aviation, Inc.

A balance housed within a hollow test body and supporting a sting for measuring aerodynamic forces. Means outside the body indicate the forces acting on the balance.

**Pressure responsive indicating means (2,785,570).** Carlyle A. Mounteer, Constant J. Chrones, Joseph F. Manildi and Walter I. Shevell, South Pasadena, Calif., assignors to G. M. Giannini & Co.

Fluid under static pressure is supplied to the exterior of two pressure-responsive capsules, and fluid under stagnation pressure is supplied to the interior of one capsule, the other being hermetically sealed. The capsules react to move an electrical switch, developing a signal in response to a predetermined Mach number, and acting in the other switch position to develop a predetermined constant signal.

**Fluid pressurizing apparatus (2,785,634).** John T. Marshall and Chas. O. Weisen-

bach, South Bend, Ind., assignors to Bendix Aviation Corp.

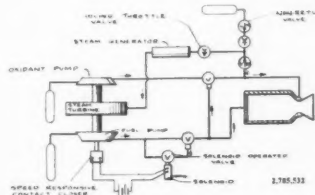
A turbine pump and a centrifugal pump are mounted on an engine-driven shaft. The turbine pump outlet is connected to the centrifugal pump inlet. A valve in a by-pass connecting the turbine pump inlet and outlet limits turbine pump discharge pressure. A fan, carried by the main shaft, cools the bearings supporting the shaft.

**Tank submerged air-driven fuel pumping system (2,785,635).** Thomas W. Johnson, Westfield, N. J., assignor to Bendix Aviation Corp.

System for supplying fuel at a proper pressure to a jet engine. Connected centrifugal pump stages, including an initial pump fuel and vapor separator stage, pump the fuel to the succeeding stages.

**Monitoring cable release mechanism (2,786,393).** David D. Grimes, Silver Spring, Md., assignor to the U. S. Navy.

Support line and release for a missile monitoring cable. An umbilical plug or connector, when released, acts to unlatch a pair of latches, permitting the cable to fall clear of the missile.



**Propellant supply systems for jet reaction motors (2,785,532).** W. Kretschmer, Southcourt, Aylesbury, England, assignor to the Government of Great Britain.

Oxidant in the form of hydrogen peroxide is fed under pressure to a steam generator supplying one or more turbines driving pumps for fuel and oxidant. For full thrust running, the oxidant is fed via a second open valve to the generator.

**Breakaway suspension band (2,786,392).** Herman Joseph Niedling, Avondale, Md., assignor to the U. S. Navy.

Apparatus for suspending and launching a missile from an aircraft bomb rack. A flexible tensioned band encircles the missile with a suspension member fixed to the upper free ends, and in lapped relationship. Tension in the band effects separation of the suspension members when the members are released from the confronting faces of an open notch on the rack.

**Process of fabricating blades for turbines, compressors and the like (2,787,049).** Edward A. Stalker, Bay City, Mich., assignor to The Stalker Development Co.

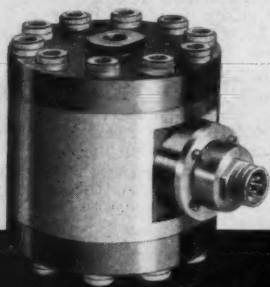
Method of placing a splined stem part in close relation with the lower wall of the blade sheet metal skin with the stem splines bearing on the inner surface of the wall. The skin is folded about the leading edge of the blade, and the parts are bonded together.

EDITORS NOTE: Patents listed above were selected from the Official Gazette of the U. S. Patent Office. Printed copies of Patents may be obtained from the Commissioner of Patents, Washington 25, D. C., at a cost of 25 cents each; design patents, 10 cents.

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# New Equipment and Processes

## Equipment

### Electrical, Electronic

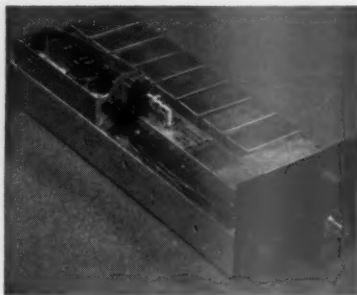
**Corrosion-Resistant Switches.** A new line of ZIAS series precision switches resist the corrosive effects of many rocket propulsion gases and the adverse conditions found in exposed locations of high speed, high altitude aircraft. Micro Switch Div., Minneapolis-Honeywell Regulator Co., Freeport, Ill.

**Missile Power Supply.** Cal-Tronics missile unit furnishes power to a missile prior to release. It plugs into the aircraft system and modifies the electrical power to proper requirements. A plastic lock-foam solidifies the unit in a metal container to withstand vibration or shock. Cal-Tronics Corp., 2211 Michigan Ave., Santa Monica, Calif.

**Power Absorber.** In use in the Redstone and Snark missiles programs, the Sun electric power absorber requires only  $\frac{1}{2}$  cu ft of space. It operates at 13,500 w continuous duty and weighs 1 lb per kw absorbed. Sun Electric Corp., 6701 S. Sepulveda Blvd., Los Angeles, Calif.

**Subminiature Terminals.** Smallest insulated terminals are claimed by Sealectro for its press-fit Teflon stand-offs and feed-thrus. They require no threads, nuts, washers, lockwashers nor seals, and withstand severest service requirements of guided missiles. Sealectro Corp., 610 Fayette Ave., Mamaroneck, N. Y.

**Recording Oscilloscope.** ETC model H-42B triggers events and records operational phenomena of various types at as many as four different points concurrently. In guided missile and other ballistics research, it is used for stress and strain analysis, resonance studies, studies of impact and vibration, and similar applications. Electronic Tube Corp., 1200 E. Mermaid Lane, Philadelphia 18, Pa.



**400-mc Receiver.** A rugged receiver developed by Bell for use in missile and guidance systems weighs 10 lb and is equally efficient as a radio-controlled receiver or as a ground communications receiver. Plug-in assemblies extend the range to 500 mc. Bell Aircraft Corp., P. O. Box 1, Buffalo 5, N. Y.

**Valve Actuator.** Electro-pneumatic valve actuators, designed to operate at variable duct pressures up to 300 lb, are used to control the air conditioning of jet aircraft cabins. Range is from -75 to 1000 F. Vapor Heating Corp., Dept. FR-57, 80 E. Jackson Blvd., Chicago 4, Ill.

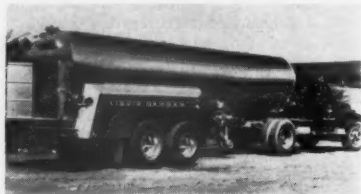
**Magnetic Shift Registers.** Individual assemblies for custom fabrication of mag-

net shift registers are produced in two types—MRC-10 (5-winding) with printed circuit base, and MRC-9 (4-winding) for use in 9-pin miniature sockets. Applications include computer storage, communication coding, data processing, time synchronizing and gating. Magnetics Research Co., 255 Grove St., White Plains, N. Y.

**Surface Temperature Transducer.** About the size of a postage stamp, the Engelhard transducer has a high thermal mass because of the platinum wire grid design, and can follow rapid temperature transients. Developed originally for use on guided missiles, it can be attached to any surface with dielectric cement. Chas. Engelhard, Inc., 850 Passaic Ave., East Newark, N. J.

### Mechanical

**Highway Transport Unit.** Liquid oxygen and nitrogen units for over-the-road use are in production in sizes from 500 to



3500 gal, built to ASME and ICC specifications. Hofman Laboratories, 219-221 Emmet St., Newark, N. J.

**Shut-Off Valve.** High performance valves for airborne use handle gases and liquids at low temperature (-320 F) and high pressure (3000 psi). Operating times from  $\frac{1}{2}$  to 5 sec are available in sizes from  $\frac{1}{4}$  to  $\frac{1}{2}$  in. Hydromatics, Inc., Cedar Grove, N. J.

**Hydraulic Pump.** Use of 66W Strato-power hydraulic pump permits utilizing systems of 5000 psi. The pump has been used for years of flight on many types of missiles, and has operated for 1000 hours of cycling endurance at maximum capacity. Watertown Div., N. Y. Air Brake Co., Watertown, N. Y.

**Reducing Regulator.** Developed for applications in missiles where regulated pressure is a prime requirement, the Val-Aero low pressure regulator is adjustable from 20 to 40 psi and accurate to better than 5 psi. Val-Aero Div., Darco Industries, Inc., 2151 E. Rosecrans Ave., El Segundo, Calif.

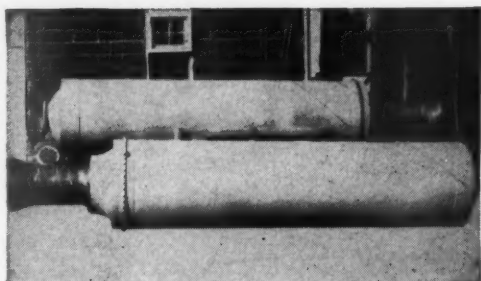
### Test

**Missile Temperature Probe.** Models T-1305 and T-1004 can be boom or strut mounted. Choice of resistance, thermistor or thermocouple elements. Range up to 1000 F with thermistor and to 2800 F with thermocouple. Aero Research Instrument Co., 315 N. Aberdeen St., Chicago 7, Ill.

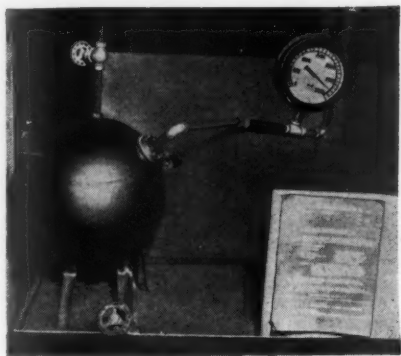
**Ballistic Missile Computer.** 101 System is completely transistorized analog computer for Glenn L. Martin ballistic missile system. Beckman Instruments, 2500 Fullerton Road, Fullerton, Calif.

**Oscillograph.** Model M-133 pen motor is designed for multichannel work. Direct inking. Response is flat from dc to

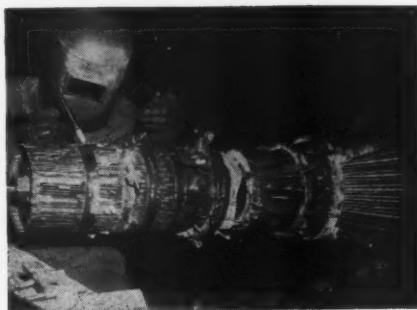
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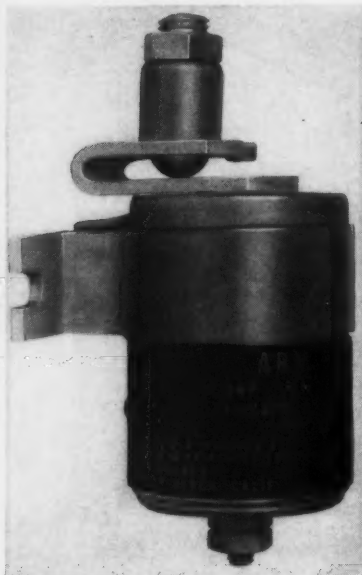
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## Missile Arming Solenoid tested at 115,000 ft.



A new arming solenoid, qualified to MIL specifications has been tested at altitudes to 115,000 ft. Designed specifically for arming of a missile on a very high performance fighter, the solenoid can be adapted to other applications. The compact unit weighs only 11 oz., has a normal power drain of 0.7 amps and operates at down to 18 volts. At  $-65^{\circ}\text{F}$  and 29 vdc power drain is only 1 amp. Operating temperature range is  $+250^{\circ}$  to  $-65^{\circ}\text{F}$ .

While specifications required testing to a lesser altitude, the solenoid was tested at 100,000' and 115,000'. At 100,000' altitude 700 RMS 60 cycle current was applied between winding and case for more than a minute. There was no corona discharge, arcing or shorting. At 115,000 feet the test was repeated with voltage being increased at a rate of 25 volts/second. At 825 volts a corona appeared suddenly. Apparently based on electron emission there was no sharply defined path of high conductance. Subsequent retesting of the identical unit showed no damage resulted from the 115,000' test.

For further information on this high altitude solenoid write to:

**STRATOS**  
A DIVISION OF FAIRCHILD  
ENGINE AND AIRPLANE CORPORATION

Western Branch: 1800 Rosecrans Ave.,  
Manhattan Beach, California.

## Tape Turns Out Missile Parts

Automation arrived in the airframe production field last month when Bendix Aircraft Co. unveiled a three-dimensional "profile" milling machine operated by a punched-tape control system.

The manufacturing operation, built specially for The Martin Co. to turn out parts for the Matador missile and the four-jet seaplane SeaMaster, first transcribes (by electronic computer) blueprint data onto a control tape. The tape is then played into a 50-ton milling machine which cuts the structural part to the desired dimensions.

A similar process, also known as "numerically controlled milling," has been used by Bendix in mass production of a small cam for jet fuel metering devices. The new development extends the tape control principle, reportedly

for the first time in this country, to large milling machines capable of turning out a whole aircraft wing section.

Bendix and Martin claim the automatic system will substantially reduce "lead time" in the production of missile and aircraft components by eliminating many weeks of tool-setup and tool-change time. Hand-made shaping tools, patterns, templates, models and many parts of the prototype are not needed.

Martin's decision to get into numerically controlled milling was based, the company reports, on present trends toward greater use of machined forgings, castings and extrusions in airframe manufacturing.

Bendix says this is the first such milling arrangement wherein the milling machine was designed specially for numerical control.



Technician monitors numerically controlled milling of a structural part for Martin missile.

60 cps. Massa Laboratories, 5 Fottler Road, Hingham, Mass.

**Accelerometer Systems.** Glennite AD series high temperature systems for testing shock and vibration will operate under environmental conditions encountered in missile applications. Units are designed for continuous operation at temperatures up to  $450^{\circ}\text{F}$  with no temperature compensation and no external cooling required. Gulton Industries, Inc., 212 Durham Ave., Metuchen, N. J.

**Target Tracking.** Aeromarker is used to visually track targets, missiles, drones or aircraft at high speeds and altitudes ranging beyond 60,000 ft. It emits a flash of light and a smoke puff which is radar reflective, and, under some conditions, attractive to infrared seekers. A fixed type unit may be mounted in the missile or aircraft structure, across the diameter of the missile, in wing pods, faired sections or attached to bomb or camera pylons. Aerojet-General Corp., Azusa, Calif.

**Temperature Recorder.** The Avien

JET PROPULSION

# TECHNIQUES and DEVELOPMENTS in oscillographic recording

FROM  
**SANBORN**

## RECORDING METHOD USED IN SANBORN DIRECT WRITERS, AND A REVIEW OF THEORETICAL AND ACTUAL ERROR FACTORS

Figure 1 shows the basic scheme by which Sanborn oscillographic recording galvanometers produce graphic records of electrical signal values. If the rapid deflection action of the heated ribbon tip stylus is visualized when current flows in the coil, it can be seen that a straight line at right angles to the chart length is recorded on the chart, at the point where the chart is drawn over a knife edge. The trace, therefore, is a true rectangular co-ordinate graph.

Since this is essentially a process of expressing coil (or stylus) deflection angles in terms of distances on a chart, the trigonometry of the situation (Fig. 2) must be examined to ascertain the accuracy of the method. Initially, and when  $\theta$  is small, the tangent and the angle are almost equal numerically. The expression  $D = R \tan \theta$  can, therefore, be rewritten  $D = R\theta$  (approx.). To the extent this latter expression is true, deflection distances (rather than deflection

angles) are an accurate measure of signal values. But to determine the extent of error resulting from using this approximation, the following data have been calculated\*, using a chart width of 25 mm either side of zero ("D" in Fig. 2) and effective stylus length of 100 mm ("R" in Fig. 2) in the series expansion for the tangent func-

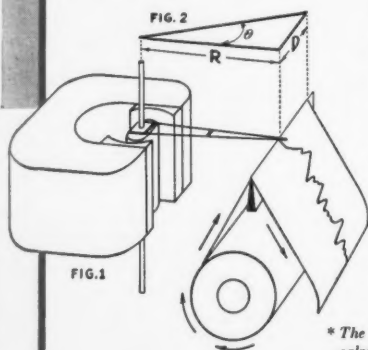
tion. Error as a function of deflection then becomes:

D mms	Radians	Theoretical Error $\epsilon$	Corrected Error $\delta$	Corrected Error in mms
10	.10	.0033	0	0
15	.15	.0075	.004	.06
20	.20	.0133	.010	.20
25	.25	.0209	.018	.45

When the recording system is calibrated, that calibration is often made on the basis of a one centimeter deflection from the chart center, or by means of a two centimeter deflection starting one centimeter below chart center and finishing one centimeter above chart center. In either case the deflection at one centimeter from chart center is accepted as the standard, and, therefore, is without error. The foregoing table can therefore be corrected by subtracting .0033 from each of the error terms to show the error,  $\delta$ , to be expected in actual use. The final column in the table shows this error in mms.

Since the active length of the stylus increases as  $\theta$  increases, deflection D increases more rapidly than  $\theta$ . All positive error terms in the series expansion bear this out, but the error terms would occur as predicted only if the galvanometer produced deflections exactly proportional to coil currents (that is, ideal spring properties in the torsion rods and uniformity of magnetic field). Pole tips in Sanborn galvanometers are proportioned so that in maximum deflections, galvanometer sensitivity decreases slightly, the compensation resulting in actual linearity better than that predicted in the table.

\* The mathematics involved here, as well as a discussion of fixed length styli, design parameters affecting over-all galvanometer performance, etc., are contained in an article by Dr. Arthur Miller "Sanborn Recording Galvanometers", published in the May 1956 Sanborn RIGHT ANGLE. Copies are available on request.



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### Rectangular Coordinates

... save analysis time, simplify interpretation and correlation of multi-channel records. No waveform curvature, negative time lines, etc.

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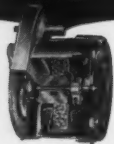
... made by hot nichrome ribbon tip of stylus on heat-sensitive Sanborn Permapaper. Clear, smudge-proof traces that clearly reveal minute signal changes.

Call on Sanborn engineers for help with your oscillographic recording application in the 0-100 cycle range. Descriptive literature is also available on request, providing data on Sanborn 1-, 2-, 4-, 6- and 8-channel Systems, choice of 12 interchangeable plug-in Preamplifiers, Separate and Supplementary Instruments.

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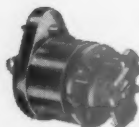
one  
source  
for all  
timers!



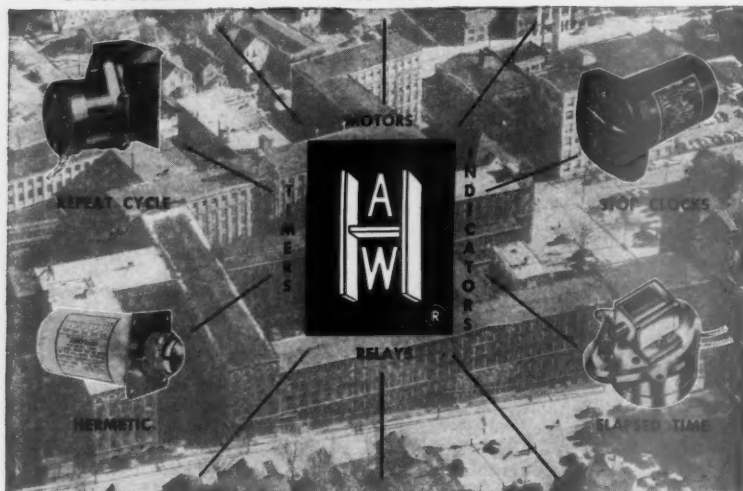
DIRECT CURRENT



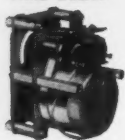
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Shown below is the new catalog of The A. W. Haydon Company describing all of the basic types of units available and many of the "specials". Included in this 25-page catalog are 60 photographs of timers, 30 dimensional drawings, and 50 charts and diagrams. This complete catalog will be supplied on request.



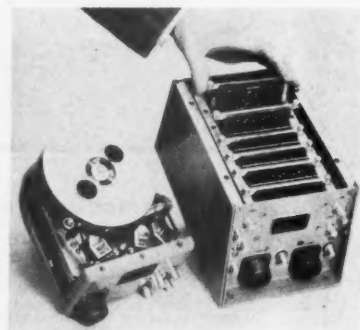
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TTR (time-temperature recorder), an overtemperature recorder for jet engines, monitors any single, group or range of engine temperatures against permissible duration time specified by the engine manufacturer. The system monitors critical temperatures within  $\pm 0.5$  per cent, and detects and logs as little as 1 sec of overheat. Avien, Inc., Woodside, N. Y.



**Midget Tape Recorder.** Missile data are automatically recorded by the model 600 automatic record-playback system for data acquisition during flight. All recording and playback electronics are on plug-in printed cards. Minneapolis-Honeywell Regulator Co., Davies Laboratories Div., 10721 Hanna St., Beltsville, Md.

## Product Literature

**Flame Plating.** Booklet gives case histories of applying tungsten carbide coatings to metals. Linde Air Products Co., 30 E. 42 St., New York 17, N. Y.

**Thermopiles.** Brochure describes models for conversion of infrared into electrical energy suitable for amplification and measurement. Jarrell-Ash Co., 26 Farwell St., Newtonville 60, Mass.

**Tungsten Carbide.** Sheet gives workability, physical and chemical properties of Kennametal Grade K501 composition with platinum binder. Kennametal Inc., Latrobe, Pa.

**Magnesium & Titanium.** Design data are given in 44-page booklet. Data covers properties, specifications, forming, etc. Brooks & Perkins, 1950 W. Fort St., Detroit 16, Mich.

**Adhesive Bonding.** Booklet illustrates bonding process for missiles and electronics industries. Metal Bonding Div., 1511 Colorado Blvd., Santa Monica, Calif.

**Photoconductors.** New bulletin presents tech specs of infrared sensitive lead sulfide cells for detection and guidance systems. Electronics Corp. of America, Photoconductor-Transistor Div., 1 Memorial Drive, Cambridge 42, Mass.

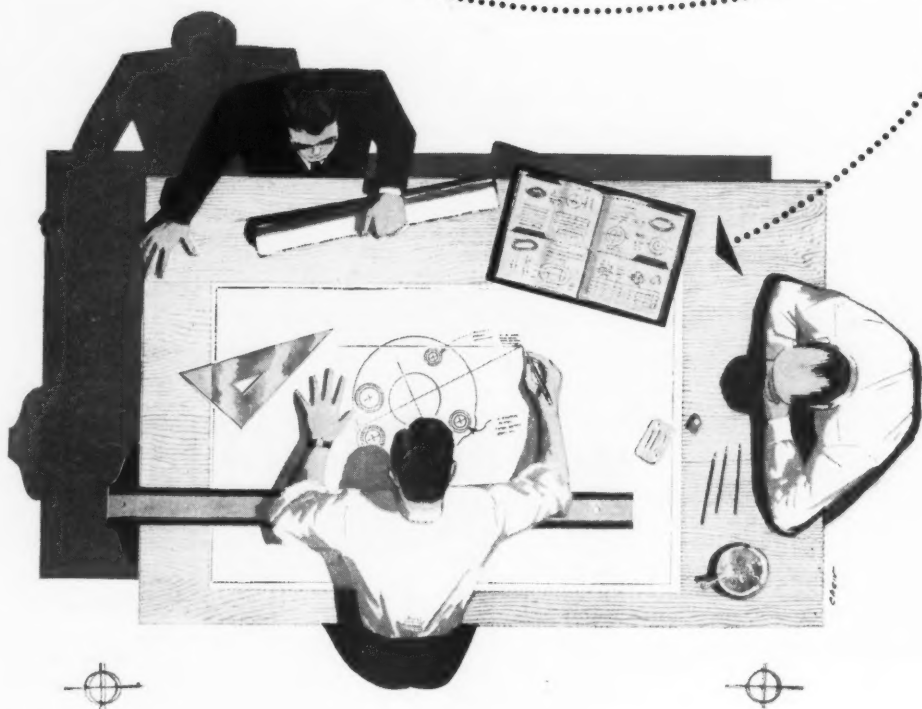
**Rocket-Powered Test Systems.** Eight-page brochure outlines use of rocket sleds for missile research and development. Hunter-Bristol Corp., Bristol, Pa.

**Silicones.** Illustrated catalog CDS-97 discusses over 115 applications for silicones. Silicone Products Dept., General Electric Co., Waterford, N. Y.

**Servo Motors.** Bulletin 385 offers information on standard and custom servo motors. Norden-Ketay Corp., Commerce Road, Stamford, Conn.

**Environmental Simulation.** Brochure illustrates various custom test equipment including chambers for Vanguard third-stage rocket. Mantee Inc., 126 Maryland St., El Segundo, Calif.

...good sealing begins  
here, too!



When the problem of sealing is a part of *design thinking* the whole design is bound to be better. This is especially true of *no-leakage* sealing. When your designs require sealing ... from  $-400^{\circ}$  to  $+1000^{\circ}$  ... why not call in one of our field men. One of the "O-seal" family\*, may be the answer to save you time, money and effort.



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# Book Reviews

Ali Bulent Cambel, Northwestern University, Associate Editor

**Numerical Methods**, by A. D. Booth, Butterworths Scientific Publications, London, 1956, vii + 195 pp. \$6.

Reviewed by PATRICK BENSON  
North American Aviation, Inc.

This book is addressed primarily to those who wish to become proficient in the art of programming automatic calculators. While one must have a thorough knowledge of the underlying mathematics to become adept at the art, the actual details of calculation are carried out automatically by machine. Since this is the case, the author judiciously avoids freighting his book with an involved presentation of such calculations—unlike most others who have written in this field. Mr. Booth's approach to his subject is straightforward, and in relatively few pages he covers a wide range of subjects.

There are three chapters of an introductory nature: One covering the subject's background, another on methods of constructing tables, and a third on methods of interpolation. Following these, the author proceeds to methods available for numerical operations with series, numerical differentiation and integration. He then treats numerical solution of ordinary and partial differential equations, systems of linear and nonlinear algebraic equations,

and integral equations. Chapters are also included on approximating functions and the numerical work involved in problems of Fourier analysis and synthesis.

The author shows an excellent understanding of modern problems in engineering and physics in his choice of topics within these subjects. Indicative of this is his chapter on Fourier analysis which is concerned largely with problems arising from its application in x-ray crystallography. Further, Mr. Booth uses recently developed methods in treating these modern problems. Among them are descent methods for the solution of linear algebraic equations, Monte Carlo methods for determining the elements of an inverse matrix, and the relaxation method as applied to the solution of equations involving the Laplace operator.

There are certain criticisms that can be leveled against the book. Exercise problems are totally lacking and relatively few problem examples are given in worked-out form. These few, however, are well done, and the examples relating to the solution of partial differential equations and integral equations are particularly timely. This lack of worked-out problems would remain a weakness even in a textbook written exclusively for the graduate-level student. Furthermore, certain chapters

could have been expanded very profitably, e.g., the chapter on approximating functions.

Speaking generally, Mr. Booth has succeeded in his primary aim—in fact, exceeded it. He has also supplied us with what can be considered the first edition of a handy reference book for the practical mathematician on numerical methods.

**A Textbook of Sound**, by A. B. Wood, 3rd edit., Macmillan Co., New York, 1955, xvi + 610 pp. \$6.75.

Reviewed by ELLSWORTH A. BROWN Jr.  
Northwestern University

This is a comprehensive book on sound including vibrations of all frequencies audible and otherwise. The book not only includes the time-honored topics and treatments but also much of the modern theory and applications of sound.

The purpose of the book as stated by the author in the preface is: "I have endeavored to write the book in a manner suited to the requirements of university students, but it is hoped that it may also prove of some value to those interested in the more technical, or applied, aspects of the subject. For those requiring further information a plentiful supply of references to original papers is provided."



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The book is arranged in five sections: I, Theory of Vibrations; II, Vibrating Systems and Sources of Sound; III, Transmission of Sound; IV, Reception, Transformation, and Measurement of Sound Energy; and V, Technical Applications.

The book discusses sound in gases, liquids and solids. It discusses sound generation, detection and measurement by mechanical, optical and electrical techniques. In particular, the author is to be commended for pointing out and carrying through the close relationship between mechanical and electrical oscillating systems.

The author has made a concerted effort to at least touch on all the topics which are related to sound. Unfortunately, the discussion of many of these topics is too brief. Further, there is a lack of up to date references on such topics.

The author's coverage of the classical topics of sound is more than adequate. This along with the wealth of information on the application and the role of the theory of sound on present-day subjects makes this book a useful addition to the library of the research worker who comes in contact with problems of sound and acoustics.

### Book Notices

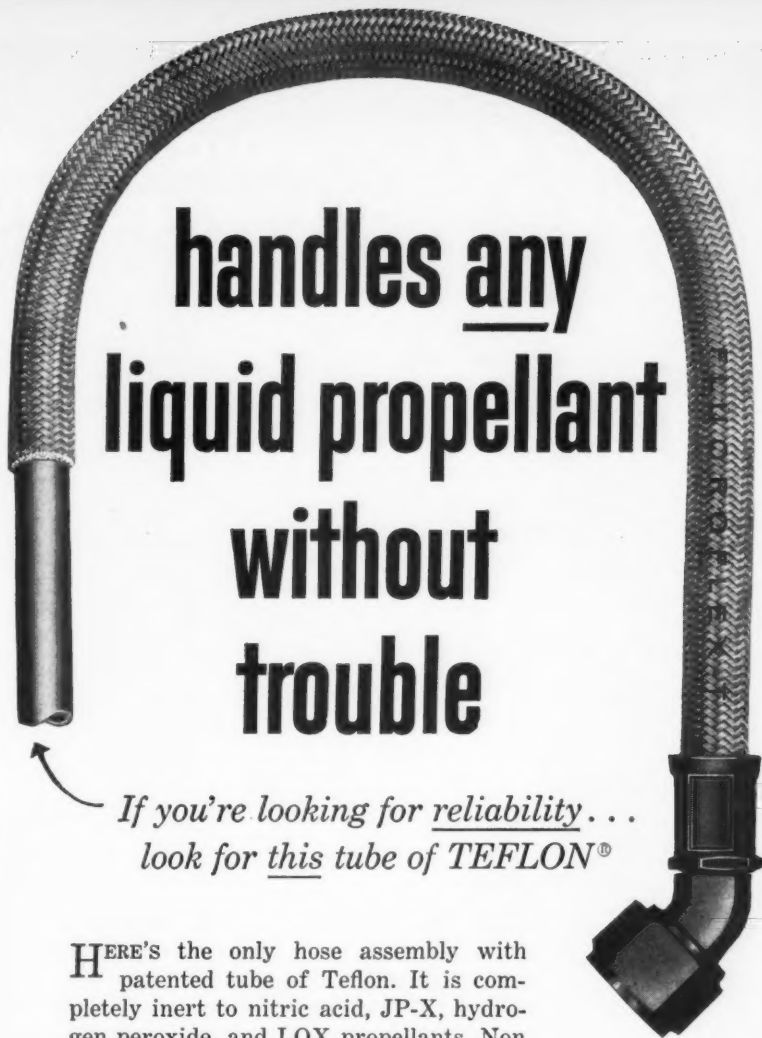
**Weather Analysis and Forecasting**, vol. II, by Sverre Pettersen, McGraw-Hill, New York, 1956, 266 pp. \$6. The second of a two-volume text on weather. In the first volume the emphasis is on the dynamics of atmospheric processes. In the second volume the emphasis is on the thermodynamics of atmospheric processes. Methods of predicting clouds and weather are presented and new techniques involving computing machines and statistical methods are discussed.

**Vapor-Liquid Equilibrium Data**, by J. C. Chu, S. L. Wang, S. L. Levy, R. Paul; J. W. Edwards, Publisher, Ann Arbor, 1956, 754 pp. \$9. A compilation of articles on vapor-liquid equilibrium data of 466 systems from 274 references up to June 1954. The data are presented in terms of mole percentage and are tabulated with temperature and pressure.

**Aircraft Today**, Edited by J. W. R. Taylor, Philosophical Library, New York, 1955, 96 pp. \$4.75. A collection of aircraft, illustrated with photographs and drawings. Aimed at the layman, it includes material of interest to men in the field as well.

**An Introduction to Matrix Methods in Theoretical and Applied Mechanics**, by S. F. Borg, J. W. Edwards, Publisher, Ann Arbor, Mich., 1956, 202 pp. \$4.75. This volume is intended as a textbook for students of engineering and applied physics. The material is treated concisely, from the viewpoint of the engineer, not the mathematician. After an introduction to the algebra and property of matrices, applications to elasticity, plasticity, fluid flow and dimensional analysis are discussed. This text should serve as a fine introduction to mathematical techniques which are imperative to the engineer who wishes to undertake modern engineering research.

**Abacs or Nomograms**, by A. Giet, translated by J. W. Head and H. D.



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**Theoretical Hydrodynamics**, by L. M. Milne-Thomson, 3rd edit., MacMillan, New York, 1955, 632 pp. \$7.50. This is the third edition of the well-known reference text. It has been rearranged and rewritten. New material has been added also. This includes: Darwin's interpretation of virtual mass, Shiffman's method of reflection across free streamlines, John's treatment of potential flow with a free surface.

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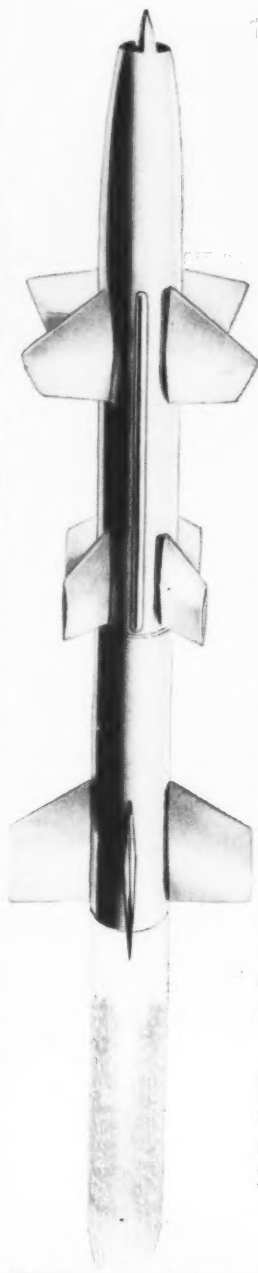
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## Jet Propulsion Engines

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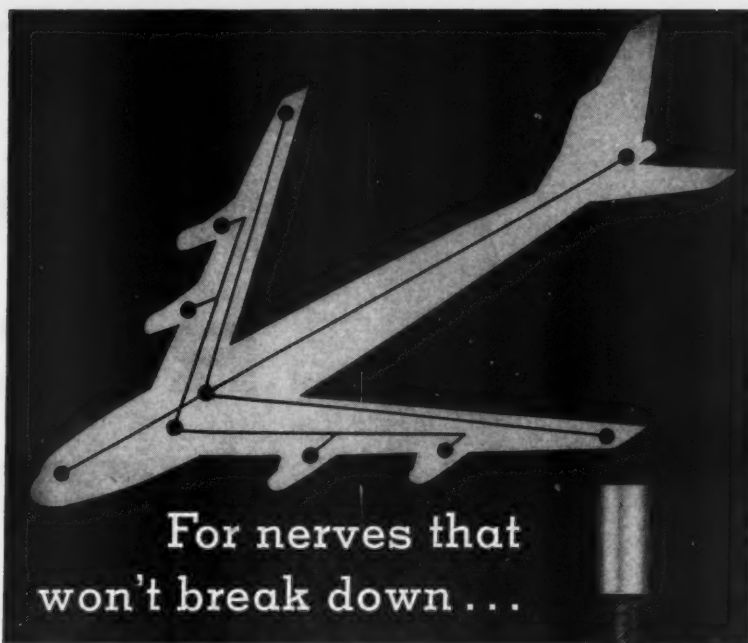
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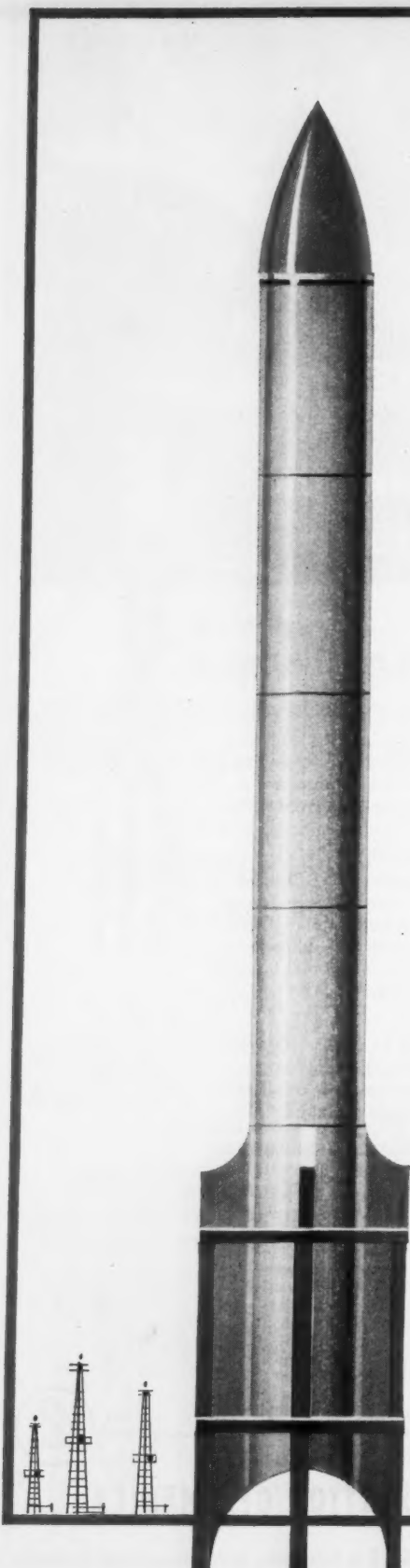


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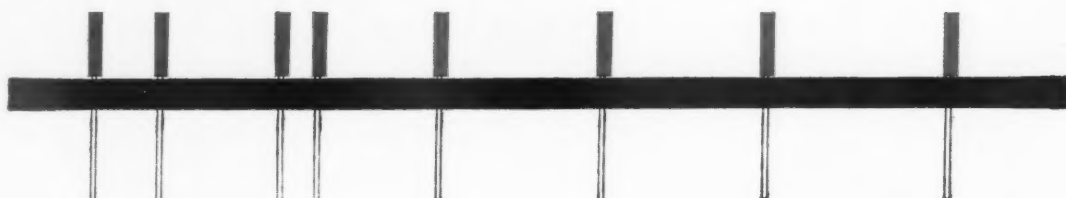
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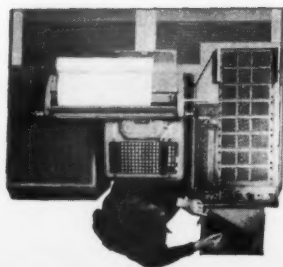
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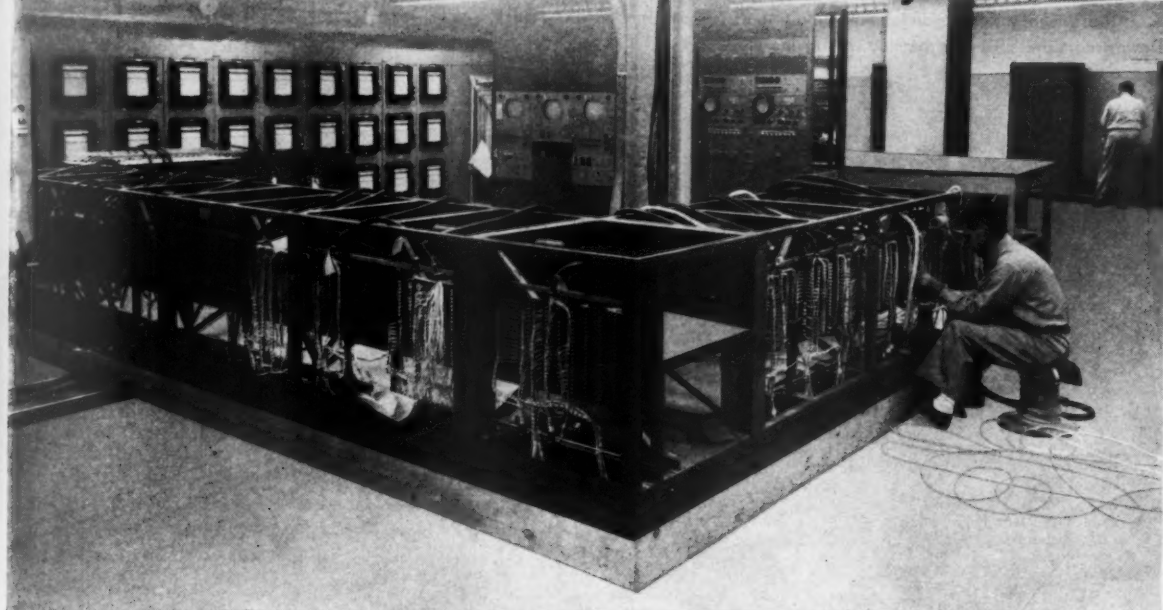
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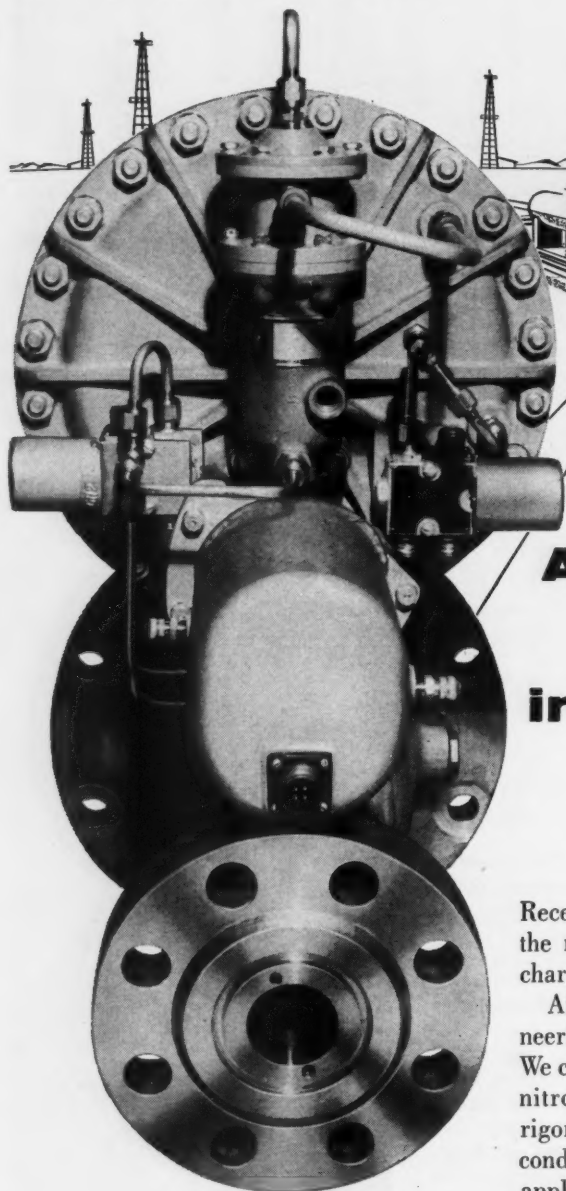
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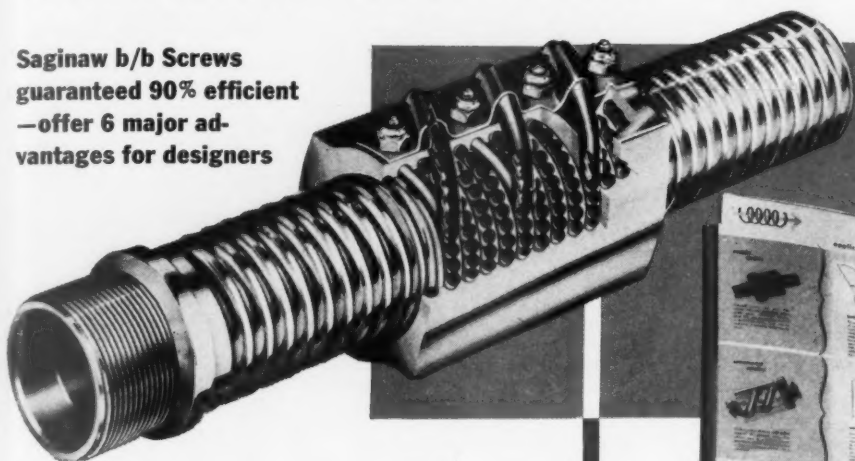
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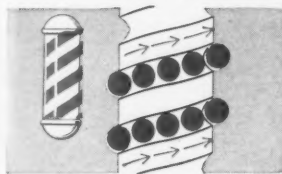
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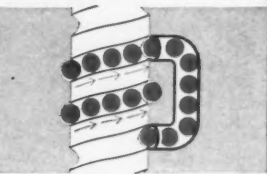


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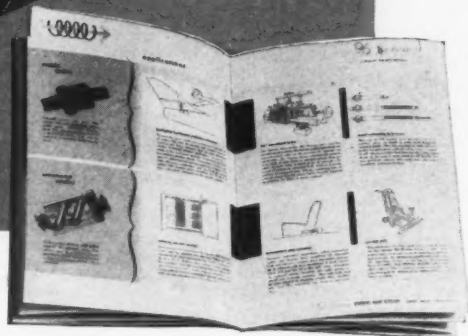


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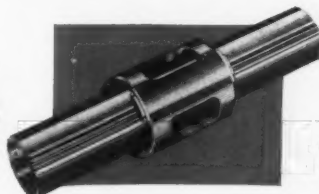


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